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INSTALLED SYSTEMS FUNCTIONAL TESTS REPORT

LIFT-FAN FLIGHT RESEARCH AIRCRAFT PROGRAM

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Installed Systems
Functional Tests Report

XV-5A Lift Fan
Flight Research Aircraft Program

September 1964

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1.0 INTRODUCTION

This report presents the results of the installed systems functional tests accomplished on the U.S. Army Model XV-5A Lift Fan Flight Research Aircraft, prior to entry into the flight test and tunnel test phases.

Testing was performed in accordance with Ryan Report No. 63B102, Installed Systems Functional Test Procedure, which is included herein as Section 7.0 and is referenced in Section 2.2.1. Changes or additions to this procedure are described in the text.

It is recognized that during the flight test and wind tunnel test phases, some systems may require modification. Whenever such modification will render obsolete any data in this report, that system will be retested to obtain data applicable to the modified system.

Results of any additional testing will be reported as addenda to this report.

2.0 SUMMARY AND REFERENCES

2.1 SUMMARY

The installed systems tests described in this report demonstrate that the aircraft systems function in accordance with design requirements. This report supplements that presented in the following Reference 2.2.2.

Test results are described in Section 3.0 of this report. Tests were divided into thirteen major functional areas as follows:

1. Electrical System Checkout
2. Surface Gains and Hysteresis
3. Flight Controls Stability
4. Flight Mode Conversion Sequence
5. Cockpit Checkout
6. Engine Run Temperature Survey
7. Engine Run Electrical System Checkout
8. Auto-Stability Tests
9. Fan Flight Trim Rates
10. Landing Gear Tests
11. Controls Proof Loads
12. Weight-Balance and Fuel Tests
13. Fire Extinguisher System Tests

Systems were modified as necessary to meet design requirements in the applicable section. Curves reflect data from final system configurations. In the case of engine runs, plotted data indicate temperature changes of selected areas from various engine operations. Complete data are retained in the Engineering Files for future reference.

2.2 REFERENCES

2.2.1 *Ryan Report 63B102, XV-5A Installed Systems Functional Test Procedure, General Electric Report No. 136.

2.2.2 Installed Systems Functional Tests Summary Report, General Electric Report No. 148.

* Included herein in the Appendix, Paragraph 7.1.

3.0 TEST RESULTS

3.1 ELECTRICAL SYSTEM CHECKOUT

3.1.1 Procedure

The electrical system checkout was performed on both aircraft per Paragraph 3.1, (see Section 7.1 of this report), of Ryan Report No. 63B102, XV-5A Installed Systems Functional Test Procedure with the following exceptions:

Step 2. Fire Extinguisher L Engine and R Engine "OFF" should be "SAFE."

Step 4. Emergency bus, essential bus and non-essential bus monitor lights come on at this step, instead of at Step 5.

Step 5. The "pressure low, boost pump" lights also come on when the annunciator test switch is pressed.

Step 21. Movement of the horizontal stabilizer and the aileron and rudder trim tabs were also monitored.

Step 22. This step was performed, following Step 24.

3.1.2 Results

The electrical systems of both aircraft functioned properly with all operations and indications occurring as specified.

3.2 SURFACE GAINS AND HYSTERESIS

3.2.1 Procedure

The surface gains and hysteresis tests were performed on both aircraft per Paragraph 3.2 (see Section 7.1 of this report). Figures 6.2.1 and 6.2.2 are schematics of the test setups.

3.2.2 Results

The results of the surface gain and hysteresis tests are presented in Figures 6.3.1 through 6.3.53. Preliminary tests of longitudinal stick position vs. longitudinal stick force in CTOL mode indicated an unbalance in breakout force. Curves are not presented for this data, since the elevator balance weights have subsequently been redistributed to eliminate the unbalance. Re-testing of stick position vs. stick force will be conducted at a later date. Calibration of the rudder pedal input displacement for the rudder hysteresis curves was done as a function of rudder pedal assembly torque tube (P/N 143K010-53) rotation. For correlation, the following figures should be used to relate to actual pedal displacement; ±28° (rudder pedal torque tube rotation) = ±12° [rudder pedal leg (P/N 143K010-11) rotation] = ±3 1/4" (rudder pedal displacement).

The characteristic curve for lateral stick position vs. R/H forward louver position at minimum collective for Aircraft No. 1, Figure 6.3.28 crosses. Also, the characteristic curve for lateral stick position vs. L/H aft louver position at maximum collective and the lateral stick position vs. R/H aft louver position at minimum collective for Aircraft No. 2, Figure 6.3.45 and 6.3.47, cross. This effect was caused by interference in the mechanical mixers. However, a modification is in progress to change the louver roll authority. This modification will require retesting of lateral stick position vs. louver position. Testing will be accomplished prior to flight in VTOL mode.

3.3 FLIGHT CONTROLS STABILITY

3.3.1 Procedure

All testing employed a mechanical sine wave generator consisting of an electronic amplifier, an amplidyne power amplifier, a variable speed DC motor (with an adjustable displacement eccentric head), and a DC tachometer generator feedback. Figure 5.3 is a photograph of this equipment. The test set-up is shown schematically in Figures 6.2.3 and 6.2.4.

Phase tests were performed on Aircraft No. 1 as outlined in Report 63B102, (Section 7.1 of this report), Paragraph 3.3 with the following exceptions:

Paragraph 3.3.1 (Report 63B102)

The lateral stick vibration was maintained at a double amplitude displacement of 0.20 inch peak to peak, from .5 cps to 30 cps, and the response curve was found using the Industrial Measurements Model 100 Servo Analyzer. As a result of preliminary tests, the aileron servo control valves were modified from an underlapped to an overlapped configuration, and the longitudinal push rods to the aileron servos were stiffened. Retests of the lateral stick to aileron (CTOL-mode) met all design requirements.

Paragraph 3.3.2 (Report 63B102)

The response from the lateral stick to the L/H forward and aft lower servos was found in two steps. For step No. 1, a response curve was found from the lateral stick to the aileron droop input push rod (stick input was 0.20 inch peak to peak). For step No. 2, a response curve was found from the aileron droop input to the L/H forward and aft louver servos. The shaker was mounted in the fuselage and was coupled to the aileron droop input push rod (aileron droop input was 0.145 inch peak to peak). This two step procedure was followed to eliminate predicted excessive controls accelerations. The two curves were combined to form an over-all response from lateral stick to louver response. As a result of initial tests, the cockpit floor was reinforced in the vicinity of the stick, and the pivot bracket for the 143C045-9 linkage at fuselage station 156.6 was reinforced. This test was repeated, with the stability characteristics meeting design requirements.

Paragraph 3.3.3 (Report 63B102)

The input amplitude excitation to the longitudinal stick was maintained at a double amplitude displacement of 0.22 inches peak to peak. The dampening ratio found at 9 cps was approxmately .0562 and at 30 cps was approximately .0281. These quantities were calculated from the frequency response curves. The dampening ratio at 30 cps is somewhat less than anticipated, however no problem will occur, because there is positive dampening.

Paragraph 3. 3. 4 (Report 63B102)

The excitation for the VTOL yaw system was applied to the rudder pedal torque tube at a double amplitude displacement of . 24 inches peak to peak. The conventional rudder control cables were disconnected aft of the forward cable tension regulator, and the rudder pedals were removed. This procedure was followed to protect the rudder and rudder pedals from excessive controls accelerations.

Initial testing had damaged the rudder pedals, which were replaced, and it was necessary to modify the yaw crank input to the mechanical mixer, to a double bearing configuration. After this modification was completed, all of the design objectives were met.

Paragraph 3. 3. 5 (Report 63B102)

The input amplitude excitation to the collective lift control was maintained at a double amplitude of 1. 0 inch peak to peak.

Paragraph 3. 3. 6 (Report 63B102)

The transient (step input) tests were conducted using a Sanborn Model 60-1300 recorder.

Paragraph 3. 3. 6. 1 Section B (Report 63B102)

Due to similarity of results, only those curves for System No. 1 are presented.

Paragraph 3. 3. 6. 1 Section C (Report 63B102)

Rudder pedal torque tube position was recorded, instead of rudder pedal position.

Paragraph 3. 3. 6. 2 Section C (Report 63B102)

Lateral stick position and aileron position vs. time was recorded. Due to good response of the lateral stick to aileron, also rudder to louvers and collective control to louvers, tests of lateral stick to louvers were not performed.

Paragraph 3. 3. 6. 2 Section D (Report 63B102)

Rudder pedal torque tube position was recorded instead of rudder pedal position.

3. 3. 2 Results

Figures 6. 3. 54 through 6. 3. 61 are the frequency response curves found during testing of Aircraft No. 1. The curves indicate that the control systems are stable. Figures 6. 3. 62 through 6. 3. 81 are the transient response time history records. The responses indicate that the controls are stable.

An investigation was made of possible pitch fan-elevator flexible body coupling problems, since the flexibility of the base of the control stick permits the elevator to excite the pitch fan modulator door system, which affects body bending and elevator position.

It was found that this flexibility caused no flexible body dynamic problems.

3. 4 FLIGHT MODE CONVERSION SEQUENCE

3. 4. 1 Procedure

The testing on both aircraft followed the procedure as outlined in Paragraph 3. 4 (see Section 7. 1 of this report), except that pitch fan inlet louver position was observed only, and the MLG was in the VTOL (wheels aft) position, with the mode change circuit disconnected.

During this test, the aircraft was operated from ground hydraulic and electrical power, or the aircraft was self-powered during engine runs and was anchored to the ground.

3. 4. 2 Results

Figures 6. 3. 82 through 6. 3. 114 are time history records of the applicable aircraft controls during normal flight mode conversions, and during abort conversions. The equivalent time history records during engine runs are also included. The time history records shown are for primary electrical power only, because operation with standby electrical power

was similar. Likewise, for single hydraulic system operation, the records for No. 2 hydraulic system only are included.

The diverter valve time as seen from Figure 6. 3. 82 was considered too short (0. 11 sec.), and it was changed to approximately 0. 30 second by adding hydraulic restrictors.

During abort tests, it was evident that the aircraft would convert from CTOL to VTOL only if the horizontal stabilizer (traveling at sufficient rate) had a minimum of 2. 5 degrees of travel before reaching the upper limit conversion stop (+7°). Thus, if a conversion is attempted with the horizontal stabilizer up higher than +4. 5°, the horizontal stabilizer would travel to +7° and stop, but no conversion would occur. Provisions are being made to allow mode conversion to be completed, regardless of horizontal stabilizer position.

During mode conversion using ground power, the velocity (rate) of the horizontal stabilizer actuator appeared sensitive to low hydraulic oil temperature. This actuator has subsequently been re-designed, and low temperature tests of the actuator and associated plumbing will be performed prior to inflight conversion.

3. 5 COCKPIT FUNCTIONAL CHECKOUT

3. 5. 1 Procedure

The tests were performed in accordance with Ryan Report 63B102 (see Section 7. 1 of this report), with the following exceptions:

Paragraph 3. 5. 1. 1 (Report 63B102)

All pressure driven instruments and switches were removed from the pitot system and the open lines capped.

Manometers and controlled pressure and vacuum sources were used for all leakage and pressure tests.

A pitot hood was used over the pitot mast so that both static and impact pressures could be applied to the pitot system simultaneously.

Paragraph 3. 5. 1. 2 (Report 63B102)

Before re-arming the pilot's seat speed sensor, the applied pressure was returned to atmospheric.

Paragraph 3. 5. 2. 1 (Report 63B102)

All static pressure driven instruments and switches were removed from the pitot system and open lines capped.

Paragraph 3. 5. 2. 2 (Report 63B102)

Before re-arming the pilot's seat speed sensor, the applied pressure was returned to atmospheric.

A functional check of the low airspeed system was made in which the flight mode was changed from CTOL to VTOL and back. The change was made at approximately 120 knots.

3. 5. 2 Results

(a) Aircraft Number 1

Landing gear warning lights and horn:	off at:	190 knots or 4200 feet
	on at:	150 knots or 3500 feet

Operation of the low airspeed indicator during VTOL mode introduced an error into the airspeed system. The discrepancy occurs as a result of the dual airspeed system, and the requirement for bleed on the pressure side of the low airspeed indicator. The error is dependent upon indicated airspeeds and varies from two knots at 100 to approximately three knots at 60. Errors below this level were not checked since it was below the range of the manometers being used. Airspeed combinations will be performed as part of the flight test program.

Pilot seat speed sensor locks out at 190 knots at sea level and 10,500 feet at zero airspeed. Upon receipt of the "Acceptance Test Standard" from Pacific Scientific Company, (Report No. IT-302, dated 2 April 1963), the speed sensor was removed from the aircraft and was tested in the laboratory to the conditions described therein.

The power quadrant and engine controls were operated and were found to function as required with the following exception: Collective throttle wind-up (engines off) appeared to be on the order of 42° twist. The mechanism has subsequently been reworked, and the wind-up reduced to a satisfactory level.

Breakout torque on collective throttle twist grip: 9.4 in. lbs. Torque required for constant turning of collective twist grip: 6.5 in. lbs. The canopy latch system and the anti-spin parachute systems functioned properly. The automatic throttle cutback system performed satisfactorily. Forces required to override the throttle cutback were 12 pounds and 14 pounds on the left hand and right hand throttles respectively.

(b) Aircraft Number 2

Landing gear warning lights and horn:	off at:	174 knots or 4500 feet
	on at:	160 knots or 4300 feet

The low airspeed system functioned as noted in (a), above; the same errors were encountered on both aircraft.

The pilot's seat speed sensor was bench tested in the laboratory to "Acceptance Test Standard" from Pacific Scientific Company as noted above.

The power quadrant and engine control were operated and found to function as required, with the exception that excessive rotational freedom was noted. The mechanism has been satisfactorily reworked. Breakout torque on collective throttle twist grip: 11.25 in. lbs. Torque required for constant turning of collective twist grip: 6.5 in. lbs.

The canopy latch system and the anti-spin parachute system and the automatic throttle cutback system performed satisfactorily. The forces required to override the throttle cutback were 14 and 19 pounds for the left hand and right hand throttles, respectively.

3.6 ENGINE RUN AND TEMPERATURE SURVEY

Engine runs were conducted on both aircraft in accordance with the procedures described below. Testing was done on Aircraft No. 2, prior to Aircraft No. 1. Each aircraft was secured to the ground as shown in Figure 5.4.

Instrumentation parameters are tabulated in Table I. 138 thermocouple measurements were continuously monitored during all engine runs. In addition, about 400 locations had temperature sensitive tabs installed. These tabs, which are visual indicators, were observed after each series of engine runs. Figure 5.5 and 5.6 are photographs of representative areas with temperature tabs. Recording instrumentation was housed in a building near the test site. Figures 5.7 through 5.10 are photographs of this instrumentation. Figure 5.11 is a photograph of the cockpit instrumentation.

Any overtemperature condition at any time during engine runs resulted in immediate shutdown, until corrective modifications were accomplished, or the temperature limits investigated and raised.

Both aircraft were equipped with sequence override diverter valve switches which enabled the pilot to convert from fan to tailpipe mode immediately without going through the entire conversion sequence.

Results of the engine runs were satisfactory unless otherwise noted.

3.6.1 Procedure and Results for Aircraft No. 2

Tests were conducted as described in the following paragraphs. Some engineering changes were incorporated to correct overtemperature conditions encountered during various phases of the engine run procedure. These changes, together with other results, are discussed in the appropriate paragraphs. All CTOL runs were made with the doors and louvers closed.

3.6.1.1 Single Engine Power Runs

Each engine was individually tested as follows:

1. The ignition switch was energized briefly and audible check of the igniter was performed.

2. The engine was then turned over by hand and checked for rubbing, dragging, or unusual sounds.

3. The air start cart was used to motor the engine, and a visual check was made of fuel flow, both at the tail pipe and the cockpit gauge.

4. The engine was run at idle rpm (48%) for approximately five minutes in both CTOL and VTOL (β_V = 0) modes for a leak check and general running inspection.

Several minor fuel and oil leaks were discovered and repaired without engineering changes. The engine was run to 99% rpm in both CTOL and VTOL (β_V = 0) modes in the following increments: 48%, 70%, 80%, 90%, 95%, and 99%. Each power setting was held until the exhaust gas temperature stabilized. Higher engine rpm's were not obtainable at this time, because of throttle linkage interference. Throttle linkage was modified to remedy this situation. Engine run and rundown time was logged throughout the tests. Figure 6.3.115 is a representative temperature/time plot of the forward landing gear door idler link.

3.6.1.2 Dual Engine Power Runs - CTOL

The engines were run simultaneously in the CTOL mode to 100% rpm in the following increments: 48%, 70%, 80%, 90%, 95%, 98%, 99%, and 100%. Each power setting was held for approximately three minutes up to 100% rpm, which was held for five minutes for heating criteria compliance.

The lower fuselage skin just aft of the tailpipe openings was overheated during sustained dual engine high power runs. The damaged skin was replaced, and the existing external insulation was extended to cover the area.

Chine rails were also installed on both sides of the aft fuselage to deflect exhaust gasses, during thrust spoiler operation, reference Figures 5.12 and 5.13.

3.6.1.3 Dual Engine Power Runs - VTOL

Power Setting	Mode	β_V	Time	
70%	VTOL	0°	2 min.	Full stroke all control combinations
70%	VTOL	-5°	2 min.	
70%	VTOL	20°	2 min.	
70%	VTOL	40°	2 min.	
90%	VTOL	0°	2 min.	
90%	VTOL	20°	2 min.	
90%	VTOL	30°	2 min.	
90%	VTOL	40°	2 min.	
95%	VTOL	0°	2 min.	
95%	VTOL	20°	2 min.	
95%	VTOL	30°	2 min.	
95%	VTOL	40°	2 min.	
100%	VTOL	0°	1/2 min.	
100%	VTOL	20°	1 min.	
100%	VTOL	30°	1/2 min.	
100%	VTOL	40°	1 min.	

During dual engine operation at idle in VTOL and $\beta_V = 0°$, a fire occurred in the crossover duct insulation just below the diverter valves. Both engines experienced over-temperature condition during this run and, consequently, were pulled and inspected.

Due to this condition, the engines will not be run under 70% rpm at any time in the VTOL mode.

As β_V was increased during subsequent VTOL runs, high temperatures were experienced in the main wheel well area and landing gear struts. The main landing gear was then wrapped with insulation and the wheel well area was enclosed. No further over-temperatures occurred in these areas. See Figures 6. 3. 115 through 6. 3. 119 for plots of overtemperature conditions in these areas.

Some pitch fan scroll leakage was evidenced. This condition was corrected by installation of a seal and baffle strip on the fan scroll. Additional insulation was also installed in the pitch fan area.

As β_V was increased in the VTOL mode, the pitch fan thrust modulator doors and inlet louvers developed an oscillation. The louver linkage was modified and dampers were installed on the thrust modulator doors to eliminate the oscillation.

Access doors were added in the engine cover and canoe for fire protection and inspection purposes.

The aluminum tubing from the dorsal fuel tank between the engines was replaced with steel tubing as suggested by the flight safety team.

3. 6. 1. 4 Thrust Spoiler Test - Dual Engine

The thrust spoiler test was run in CTOL mode with both engines as follows:

Power Setting	Spoiler Position	Time
90% rpm	12-1/2%	30 sec.
	25%	
	37-1/2%	
	50%	
90% rpm	62-1/2%	30 sec.
70% rpm	Retracted	2 min.
95% rpm	12-1/2%	30 sec.
	25%	

Power Setting	Spoiler Position	Time
	37-1/2%	
	50%	
95% rpm	62-1/2%	30 sec.
70% rpm	Retracted	2 min.
100% rpm	25%	30 sec.
	50%	
	75%	
100% rpm	100%	30 sec.

At the conclusion of the 95% rpm runs, the titanium aft center fairing was replaced with a heavier gage material, due to cracking of the original installation.

Exhaust gasses entered the fuselage tail pipe opening around the shrouds, causing a local over-temperature condition at the aft fuselage canted bulkhead (ref. Figures 6. 3. 120 and 6. 3. 121 TS-458). Stainless steel finger seals were installed in this opening, correcting the over-temperature problem.

The fuselage Fiberglas insulation became frayed just aft of the tail pipe opening during 100% rpm and 50% spoiler runs. It was replaced by . 016 inch thick stainless steel over Min-K insulation.

3. 6. 1. 5 Single Engine Diverter Valve Tests

This test was run as follows:

Mode	L/H Eng. RPM	R/H Eng. RPM
CTOL	90%	70%
VTOL	90%	70%
CTOL	95%	70%
VTOL	95%	70%
CTOL	100%	70%

Mode	L/H Eng. RPM	R/H Eng. RPM
VTOL	100%	70%
CTOL	70%	100%
VTOL	70%	100%

3.6.1.6 Pitch Fan Thrust Modulator Door Test

With both engines at 100% rpm in VTOL mode and $\beta_V = 0°$, the control stick was moved rapidly as follows: From extreme forward and aft positions to neutral: from half forward and aft positions to stop and neutral (both directions from each position). Some pitch fan thrust modulator door vibration was experienced while at the aft stick positions, which was eliminated by installation of dampeners. This test was run with pitch fan inlet louvers temporarily removed.

3.6.1.7 Flight Mode Conversion

Diverter valve electrical system was restored to original configuration with the exception of the landing gear mode change actuator valve, which remained de-activated. The automatic longitudinal trim switches were de-activated during this test due to a malfunction caused by switch cams being forced off their tracks during full forward or aft stick movement. Simulator tests indicated that these switches were unnecessary, and they were deleted. An auto-trim signal has subsequently been incorporated to maintain stabilizer in "full-up" position during VTOL mode. Conversions were made from CTOL to VTOL and from VTOL to CTOL at 70% and 90% engine rpm. Aborts from CTOL to VTOL to CTOL, and from VTOL to CTOL to VTOL were also accomplished at 90% engine rpm.

3.6.1.8 Fan Overspeed Cutback

Performance of the fan overspeed cutback system was satisfactory in all respects.

3.6.2 Procedure and Results for Aircraft No. 1

This series of tests was conducted as described in the following paragraphs. All engineering changes resulting from

engine runs on Aircraft No. 2 were incorporated into Aircraft No. 1 prior to testing. The only modification to Aircraft No. 1 during testing was to the pitch fan inlet louvers. These louvers were unstable in the VTOL mode at low β_V angles. The louver actuator mechanism pre-load was increased, and the louver hinge structure was stiffened by the addition of aluminum saddles bonded to the Fiberglas hinge support beam. This problem was discovered during Aircraft No. 2 engine runs. However, due to time limitations, it was decided to study the problem during testing of Aircraft No. 1. The modification was accomplished at that time. All CTOL runs were made with the fan doors and louvers open.

3.6.2.1 Single Engine Power Runs

Both engines were run individually as follows:

1. The ignition switch was energized briefly, and an audible check of the igniter was performed.

2. The engine was then turned over by hand and checked for rubbing, dragging, or unusual sounds.

3. The air start cart was used to motor the engine, and a visual check was made of fuel flow both at the tail pipe and cockpit gauge.

4. The engine was run at idle rpm for approximately five minutes in both CTOL and VTOL (β_V = 0) modes for a leak check and general running inspection. Several minor fuel and oil leaks were discovered and repaired without engineering changes. The engine was run to 100% rpm in both CTOL and VTOL (β_V = 0) modes in the following increments: 48%, 70%, 80%, 90%, 95%, 98%, 99% and 100%. Each power setting was held until the exhaust gas temperature stabilized. Engine rundown time after shut-off was logged throughout the tests.

3.6.2.2 Dual Engine Power Runs - CTOL

The engines were run simultaneously in the CTOL mode to 100% rpm in the following increments: 70%, 90%, 95%, and

100%. Each power setting was held for approximately two minutes.

3.6.2.3 Dual Engine Power Runs - VTOL

The dual engine VTOL runs were conducted according to the following schedule, with returns to 70% engine rpm in CTOL for cooling when structural overheat light in pilot's instrument panel came on. False indication of overheat was caused by incorrect resistance thresholds. This condition has subsequently been corrected.

Power Setting	Mode	β_V	Time	
70%	VTOL	0	2 min.	Full stroke all control combinations
70%	VTOL	-5°	2 min.	
70%		20°	2 min.	
70%		40°	2 min.	
90%		0	2 min.	
90%		20°	2 min.	
95%		0	2 min.	
100%		0	1/2 min.	
95%		20°	1 min.	
95%		40°	1 min.	
*98%		20°	1 min.	
*98%		40°	1 min.	

*100% engine rpm not obtainable at this time due to automatic fan overspeed cutback.

3.6.2.4 Thrust Spoiler Test - Dual Engine

With both engines at 90% rpm in CTOL mode, the spoilers were run through their full travel at maximum rate.

3.6.2.5 Pitch Fan Thrust Modulator Door Test

With both engines running at 100% rpm in VTOL mode and with $\beta_V = 0°$, the control stick was moved rapidly as follows: from extreme forward and aft positions to neutral: from half-forward and half-aft positions to stop and neutral (both directions from each position).

3.6.2.6 Flight Mode Conversion Test

The diverter valve electrical system was restored to original configuration with the exception of the landing gear mode change actuator valve which remained de-activated. Conversions were made from CTOL to VTOL, and from VTOL to CTOL at 90% engine rpm. Aborts from CTOL to VTOL to CTOL, and from VTOL to CTOL to VTOL, were also accomplished at 90% engine rpm.

3.7 ENGINE RUN ELECTRICAL SYSTEM CHECKOUT

3.7.1 Procedure

The electrical system checkout was performed on both aircraft in accordance with Ryan Report 63B102, (see Section 7.1 of this report, Paragraph 3.7), with the exception of the battery test, which will be performed at a later date.

3.7.2 Results

Two diodes were added to each generator control panel. This was necessary to accomplish the switch over from battery to generator without losing the emergency bus in the process. Some difficulty was encountered in the load sharing function between the two generators, but this was corrected by adjustment.

3.8 AUTO-STABILITY SYSTEM CHECKOUT

3.8.1 Procedure

The auto-stab system checkout was performed on both aircraft per Paragraph 3.8 of Ryan Report 63B102, (see Section 7.1 of this report). Figure 6.2.6 is a schematic

drawing of the test setup. In addition to the checks described, frequency response tests of the system were made for the following conditions:

1. Aircraft No. 1, Pitch Fan Thrust Modulator Door Servo Response - with door dampener, without notch network, without simulated vane inertia.

2. Aircraft No. 1, Pitch Fan Thrust Modulator Door Servo Response - with door dampener, with notch network, without simulated vane inertia.

3. Aircraft No. 1, Pitch Fan Thrust Modulator Door Servo Response - with door dampener, with notch network, with simulated vane inertia.

4. Aircraft No. 2, Left Hand Forward Wing Fan Exit Louver Servo Response - yaw input, without notch network.

5. Aircraft No. 2, Left Hand Aft Wing Fan Exit Louver Servo Response - Yaw input, without notch network.

6. Aircraft No. 2, Left Hand Forward Wing Fan Exit Louver Servo Response - roll input, without notch network.

3.8.2 Results

The auto-stab systems of both aircraft functioned properly with all operations and indications occurring as specified. The results of the frequency response tests are shown in Figures 6.3.122 through 6.3.127. Comparison of Figure 6.3.122 and 6.3.123 show the effect of the amplifier notch network on the pitch servo response. Comparison of Figures 6.3.123 and 6.3.124 show that in the latest configuration, the addition of a mass to simulate the added inertia of the turning vane on the pitch fan thrust modulator door had no significant effect on the frequency response. Figures 6.3.125 through 6.3.127 show the response of the wing fan exit louver servos to roll and yaw inputs, which is the same for either roll or yaw signals from the amplifier.

3.9 FAN FLIGHT TRIM RATES

3.9.1 Procedure

These tests were performed on Aircraft No. 2 per Paragraph 3.9, Ryan Report 63B102, (see Section 7.1 of this report), with the following exceptions:

Paragraph 3.9 (Report 63B102)

The main landing gear was in the VTOL (wheels aft) position, the landing gear STOL-NORM switch was in the normal position, and the landing gear mode change electro-hydraulic control valve was disconnected electrically.

Paragraph 3.9A (Report 63B102)

With the vector actuator at $\beta_V = 0°$, the pitch trim was energized through a complete stroke. A record of pitch fan door servo position vs. time was made. The vector actuator was energized to $\beta_V = 15°$, then the pitch trim and thrust vector actuator were both energized, while a record of pitch control door servo position, thrust vector actuator position and horizontal stabilizer position vs. time was made.

Paragraph 3.9B (Report 63B102)

With the vector actuator at $\beta_V = 30°$, the pitch trim was operated through full horizontal stabilizer stroke. A record of horizontal stabilizer position vs. time was made.

Paragraph 3.9C (Report 63B102)

With the vector actuator at $\beta_V = 0°$, the roll trim was energized through a complete stroke. A record of louver servos position vs. time was made.

Paragraph 3.9D (Report 63B102)

With the vector actuator at 50°, (full closed), the LH aileron tab trim rate was checked using a protractor and a stop watch.

Paragraph 3.9F (Report 63B102)

With the vector actuator at 50° (full closed), the rudder tab rate was checked using a protractor and a stop watch.

3.9.2 Results

The following are trim rates found as a result of this test:

CONTROL	β_V		
Pitch Fan Thrust Modulator Door	0°		2.39 Deg/Sec
Horizontal Stabilizer	48°	Emergency Trim	1.085 Deg/Sec
Horizontal Stabilizer	30°		3.98 Deg/Sec
Horizontal Stabilizer	Full Closed		1.15 Deg/Sec
Vector Actuator	0		3.79 Deg/Sec
Roll Trim	0	Left Wing	0.652 B_s Deg/Sec
Roll Trim	0	Right Wing	0.719 B_s Deg/Sec
Yaw Trim	0	Left Wing	0.949 B_s Deg/Sec
Yaw Trim	0	Right Wing	0.855 B_s Deg/Sec
Roll Trim	Full Closed		6 Degs/8.5 Sec (Tab)
Yaw Trim	Full Closed		22 Degs/10.5 Sec (Tab)

Trim Control Transfer Point β_V = 16°

3.10 LANDING GEAR FUNCTIONAL TESTS

3.10.1 Procedures

Testing was performed on Aircraft No. 1 in accordance with Paragraph 3.10, Ryan Report 63B102, (see Section 7.1 of this report), except as follows:

A retraction time test under simulated 1-1/2 g load factor, in addition to conducting a check of hydraulic systems back pressure during control cycling, was performed. The additional load was applied to the main landing gear by distributing lead-shot bags on the shock strut. The system back pressure checks were made by placing a hydraulic gauge in the return line adjacent to the landing gear up-locks, and noting the pressures as the control system was cycled.

Paragraph 3.10.6.1 (Report 63B102)

The nitrogen system pressure was lowered from 3000 psig to 2250 psig.

Paragraphs 3.10.7, 3.10.8, 3.10.9, 3.10.10 and 3.10.11 (Report 63B102)

These tests were not accomplished due to modification of landing gear (heat shielding) for engine runs. Tests will be accomplished during flight test phase prior to operational use of system.

3.10.2 Results

Brake Check

Brake assembly dimensions were checked per Goodyear Service Bulletin (GTR/UP 1) entitled "Maintenance and Overhaul Instructions for Goodyear Wheels and Brakes used on Ryan XV-5A", dated May 1963.

Distance between brake disc and face of lining: .005/inch minimum, both wheels. Distance between brake disc and brake assembly: .22 inch, both wheels.

CTOL Main Landing Gear Functional Test

The aircraft was placed on jacks for this and all of the following landing gear checks:

The following actuation times were noted:

CTOL to VTOL position: 6.6 seconds

VTOL to up and lock (doors closed): 4.4 seconds

Total time from down and locked CTOL position to up and lock: 11 seconds

Up and locked to down and locked, CTOL position: 12.5 seconds

Operation of all warning and position indicators was satisfactory.

VTOL Main Landing Gear Functional Test

Operation of the landing gear in this mode was normal. Retraction time was 3.5 seconds.

Downlock Override Check

The main landing gear wheels were placed on wooden blocks high enough to release the microswitches on the torque links. The blocks were placed under each individual wheel and then under both wheels. In each case, the override button above the landing gear selector switch had to be used to retract the gear.

1-1/2 g Time to Retract

This particular portion of the functional test was added to insure proper in-flight retraction. Fifty pounds of lead shot bags were placed in the vicinity of each main landing gear axle. The time required to retract was 3.7 seconds.

STOL Override

Operation of the landing gear in the override condition was normal. The landing gear was retracted and extended while

in the override position. Mode changes were also accomplished with no discrepancies.

Emergency Pneumatic Extension System

No-Load Functional Test

A change to the test procedure was made whereby the system pressure was reduced from 3000 to 2250 psig. The orifice size in the -11 restrictor in the nitrogen system was increased from the original .0105 inch diameter to .0156 inch diameter. After this modification, the following was noted:

System pressure: 2200 psig

Doors start to open to doors fully open: 2.7 seconds

Doors start to open to landing gear unlatch: 4.2 seconds

Doors start to open to landing gear down and lock: 12.9 seconds

1875 psig nitrogen remaining

Approximately 4.5 minutes were required to bleed the hydraulic system of the nitrogen after the emergency extension handle was returned to the normal position.

Low Pressure Emergency Extension with Simulated 180 Knots Air Drag

A cable and pulley system was erected which applied 175 pounds rearward force (drag simulation) to the landing gear at the uplock lug. The cable was so rigged that the full load was applied during the last 30° of gear travel. The nitrogen pressure was then reduced to 1700 psig for the low pressure test. (It was noted that the low pressure light came on as soon as the emergency extension handle was pulled.) The emergency extension under load provided the following data:

Nitrogen System Pressure: 1700 psig

Doors start to open to doors fully open: 2.7 seconds

Doors start to open to landing gear unlatch: 5.8 seconds

Doors start to open to landing gear down and lock:
29 seconds

1500 psig nitrogen pressure remaining.

The landing gear and door actuation was smooth, and good rates were maintained throughout the gear extension phase. Landing gear down locking was positive. It was noted that with an increase in simulated drag of approximately 11.4%, the main landing gear would not fully extend.

Hydraulic System Back Pressure Test

During the installation and checkout phase of the landing gear, and prior to the functional tests, problems had been encountered with surges in the hydraulic return line pressures. The surges resulted in premature up-lock release during gear extensions. In order to simulate actual system back pressures while using an external hydraulic supply, a valve was placed in the return line and a back pressure of 80 psig was created at the MLG locations.

It was previously found, through the use of a manual hydraulic pump, that pressures of 150 psig would release the up-locks on the main landing gear. Subsequent to these tests, a check valve was installed in the return line to preclude any surges releasing up-locks prematurely.

During the time, the constant 80 psig system back pressure was being applied at the MLG, the control systems were cycled continuously for several minutes. Surges in back pressure were checked; build-ups to approximately 95 psig were noted.

During the functional checks, it was found that the relief valve on the Number 1 hydraulic system reservoir opened at 100 psig on the return line.

3.11 CONTROLS PROOF LOADS

3.11.1 Procedure

The controls proof test was run on Aircraft No. 2 in accordance with Paragraph 3.11, Ryan Report 63B102 (see Section 7.1 of this report), with the following exceptions:

Paragraph 3.11.1 (Report 63B102)

CTOL Mode

a. The aileron surfaces were restrained by pieces of 1/2-inch plywood clamped to the trailing edge of the wing and aileron.

b. Lateral stick force of 100 pounds was manually applied using a spring scale.

c. A Bimba-type hydraulic cylinder was used to apply rudder pedal and longitudinal stick forces of 300 and 200 pounds respectively.

d. Forces were manually recorded from a hydraulic gauge and spring scale.

e. The collective control stick was loaded to 150 pounds in both up and down directions using a spring scale and dead weight.

f. A steel scale was used to record deflections.

Paragraph 3.11.2 (Report 63B102)

VTOL Mode

a. With the aircraft in VTOL mode, the control stick and both rudder pedals were displaced to their extreme positions (separately) with hydraulic power on, and held firmly as hydraulic power was shut off. A spring scale was used to pull the stick or rudder pedal in the opposite direction from which it was displaced. The force required to bring it to the cockpit stop was recorded.

b. In VTOL mode, the control stick was restrained in the full aft position by a 100-pound force. With the horizontal stabilizer in the leading edge full-down position, an increasing down load was applied to the elevators until the control stick just cleared the aft stop.

c. Throttle testing was included in addition to the above tests. A 75-pound load was applied to both throttles

(individually) with the load reacted by bottoming of the lower throttle mechanism. Deflections were measured and recorded.

3.11.2 Results

The elevator aileron and rudder control systems and collective control stick successfully withstood the applied pilot effort loads. During lateral stick loading, the control stick pivot tube pulled out of its aft bearing support. A repair was made and the load was re-applied successfully. The aileron, elevator and rudder control system deflection data appear in Figures 6. 3. 128 through 6. 3. 131.

In the VTOL mode tests, the following forces were recorded:

Stick - 35 pounds to lateral stops
5 pounds to longitudinal stops

Rudder Pedals - 88. 4 pounds to L/H stop
132 pounds to R/H stop

The elevator moved down 6° before first movement of the stick occurred. The outboard throttle handle deflected 5/8 inch aft, and the pivot 1/8 inch downward. The inboard throttle handle deflected 1/8 inch aft with no noticeable movement of the pivot. Subsequent modifications to the throttle quadrant have been accomplished to reduce deflections.

3.12 WEIGHT AND BALANCE

3.12.1 Procedure

The aircraft weight and balance tests were performed on Aircraft No. 1 in accordance with Paragraph 3. 12, Ryan Report 63B102 (see Section 7. 1 of this report), except as follows:

Aircraft attitudes varied somewhat from those shown in the test procedures report, and are noted on each series of curves.

Centers of gravity were not defined in sufficient detail when the individual fuel tanks were installed in the aircraft, so the aft main and dorsal fuel tanks were removed and tested individually. The tanks were set up in fixtures which allowed them to be rotated while on the weighing scales. Figures 5.14 and 5.15 are photographs of Aircraft No. 1 being weighed.

3.12.2 Results

A survey of installed equipment showed the following items to be missing:

1. Fibreglas flap hinge fairings
2. Throttle quadrant cover
3. Annunciator panel
4. Signal conditioner box
5. Telemetry box
6. Seat mode speed sensor
7. Engine compartment cooling fans and gear boxes (20 pounds of lead shot were added at each fan and gear box location to simulate component weights)
8. Heat shield modification for main landing gear
9. Chine rail installation above tail pipe exits and modification to original Min-K insulation installation in the tailpipe area
10. Pilot seat rocket motor

The net weight, not including the above items, was 7874 pounds. The center of gravity of the aircraft in the level attitude at this weight was located at F.S. 245.52 and W.L. 117.48.

The following fuel quantities were required to fill the tanks indicated:

Forward fuselage tank:	1600 pounds
Aft main tank:	830 pounds*
Dorsal tank:	797 pounds

*Slightly more fuel may be added to this tank when installed in the aircraft, as it will be filled through the dorsal tank, and may be filled completely. During testing, the tank was filled through its own filler, which will not allow the tank to become completely filled.

Cockpit fuel quantity gauges showed some discrepancy throughout the fuel tests, and therefore are not shown. The error in the forward tanks was due to a calibration error which was later corrected. The aft and dorsal tank fuel quantity transmitters malfunctioned. These transmitters were re-worked and field tested prior to first flight. All fuel quantity gauges now indicate fuel quantities within the design tolerances.

Curves showing fuel center of gravity position as a function of fuel quantity are presented in Figures 6. 3. 132 through 6. 3. 137.

Table II shows unusable fuel as a function of aircraft attitude.

3. 13 FIRE EXTINGUISHER SYSTEM

3. 13. 1 Procedure

The extinguisher system was tested by discharging a bottle into the closed engine compartment and analyzing dynamically the resulting atmosphere within the compartment. An FAA-furnished gas analyzer system was used under the supervision of an FAA representative, and results were recorded on a CEC oscillograph. Photographs and a diagram of the test setup appear in Figures 5. 16 and 6. 2. 7 respectively.

3. 13. 2 Results

The fire extinguisher system provided a concentration of mono-bromo-tri-fluoromethane ($CBrF_3$) that was more than

adequate for ground fire extinguishing. It was recommended by the FAA representative that further tests be made in flight.

4.0 TEST EQUIPMENT

Name	Model	Accuracy	Use
Linear Potentiometers	Mektron C0080, C00803, C0084	.5%	Surface & Controls Position
DC Power Supply	Hewlett Packard 721A	1%	Potentiometer Excitation
Oscilloscope	Dumont 304	-	Lissajous Patterns - Frequency & Vibration
X-Y Plotter	Moseley 135	3%	Hysteresis-Gains and Frequency Response
Oscillograph	Sanborn 60-1300	3%	Surface Position vs. Time
Visicorder	Minneapolis-Honeywell 906A	2%	Automatic Functions & Surface Positions
Servoscope	Servo Corp. of America 1100 c	-	400 cps Carrier Modulator
VTVM	Ballantine 300 D	2%	Voltage
Timer	Minneapolis-Honeywell 906 C	2%	Visicorder Timing
Servo Analyzer	Industrial Measurements 100	phase 3% Freq. 2% amp. ratio .4DB	Phase Shift & Amp. Ratio

Name	Model	Accuracy	Use
Mechanical Sine Wave Exciter	General Dynamics	-	Vibrate Controls
Air Start Cart	General Electrical MA-1A	-	Start Engines
Visicorder	Minneapolis-Honeywell 1508 945 A	2%	Position, RPM, Etc.
Oscillograph	Consolidated Electrodynamics P4-114	3%	Load Cell
Amplifiers	Consolidated Electrodynamics System D	-	Signal Amplification
Power Supply	Perkins MTR 636-5D Dressen Barnes 62-121	3%	Instrumentation Excitation
VTVM	B & K 2409	3%	Vibration
E Put Meter	Hewlett Packard 522B	.005%	Fan RPM
Load Cell	Baldwin Lima U-1	.5%	Lift

Name	Model	Accuracy	Use
Brown Recorders	Minneapolis-Honeywell 161632; SY153 x 66 - (PSD12) II-III 50; 153 x 65P16 II-III 81; Y153 x 66 - (PSD12) x 50 (v) Daystrom-Weston 6702	±2 Degrees	Temperatures
Power Supply	Nobatron NA 28-125	3%	External Power
Spring Scales	Fairbanks 0-150 lb; 0-50 lb.	± 1/4 lb.	Observe and Apply Stick Forces
Hydraulic Cylinder	Bimba 046-DX	-	Apply Stick and Rudder Forces
Manometer	Meriam 31EC10; 33KB35	3%	Pressure
Pitot Hood	Ryan 124M103	-	Apply Pressure
Scales	Fairbanks 15000 lb.	± 2 lb.	Weigh Aircraft
DC Power Supply	General Electric GPD 265 AA1	-	External Power (Electric)
Hyd. Power Supply	Greer Model D-4 (Electric Motor Driven) (0-20 GPM) (0-5000 PSIG)	-	External Power (Hydraulic)

Name	Model	Accuracy	Use
Hyd. Power Supply	Greer Model PG3C-10VH (Gasoline Engine Driven) (10 GPM @ 3000 PSIG)	-	External Power (Hydraulic)

5.0 PHOTOGRAPHS

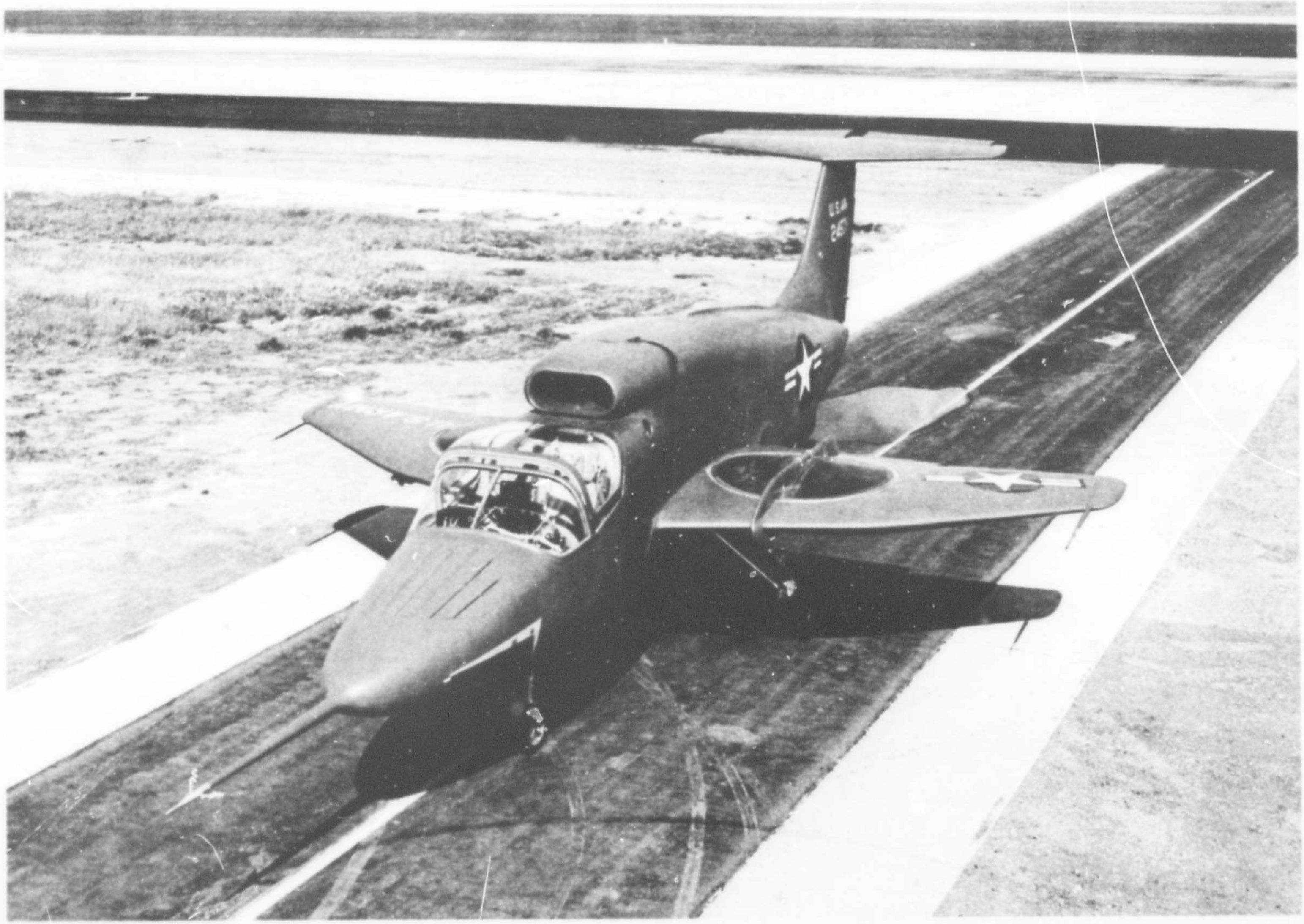


Figure 5.1 Aircraft No. 2

Figure 5.2 Aircraft No. 2

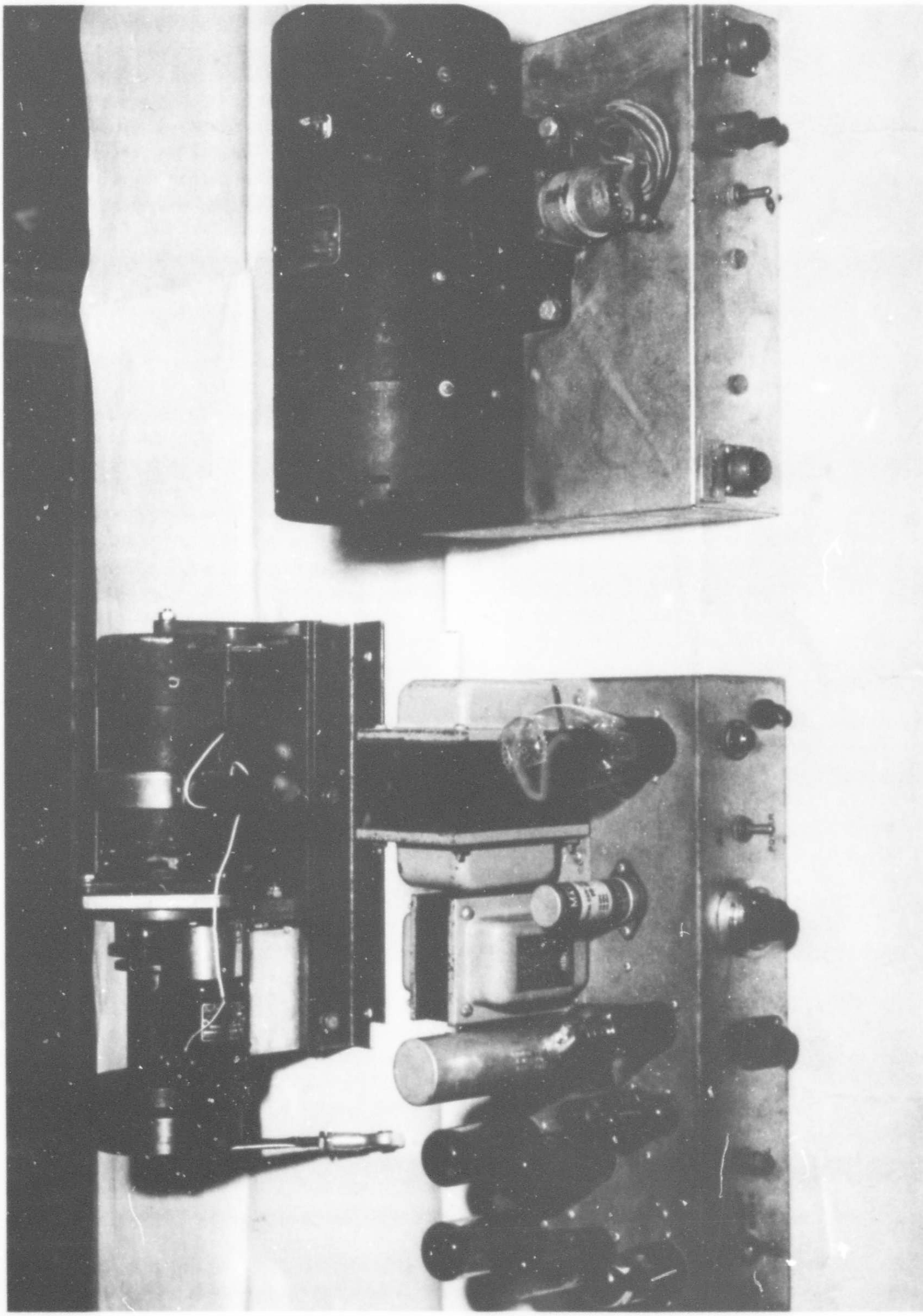


Figure 5.3 Mechanical Sine Wave Generator

Figure 5.4 Aircraft Tie-Down

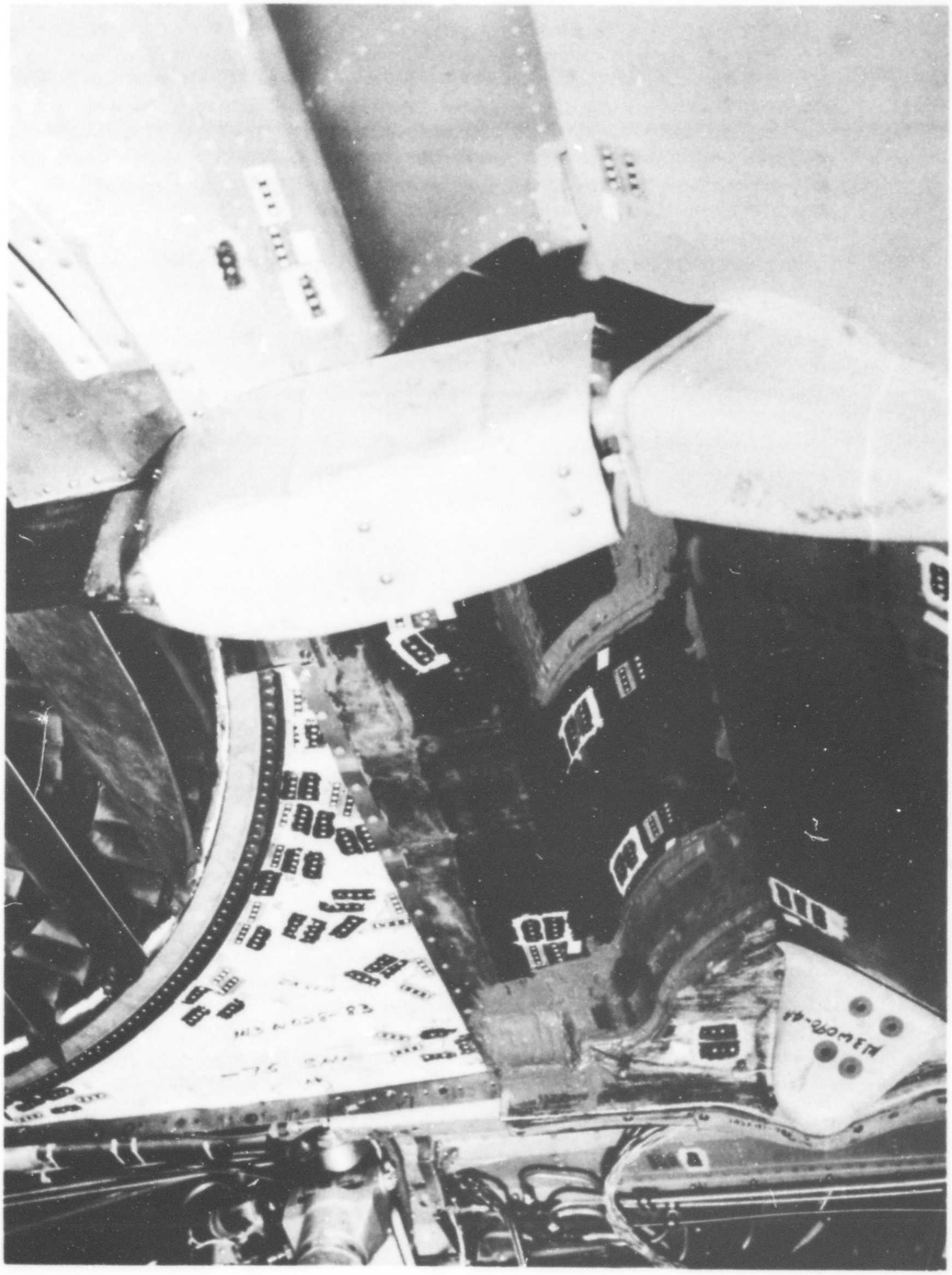


Figure 5.5 Temperature Tab Installation

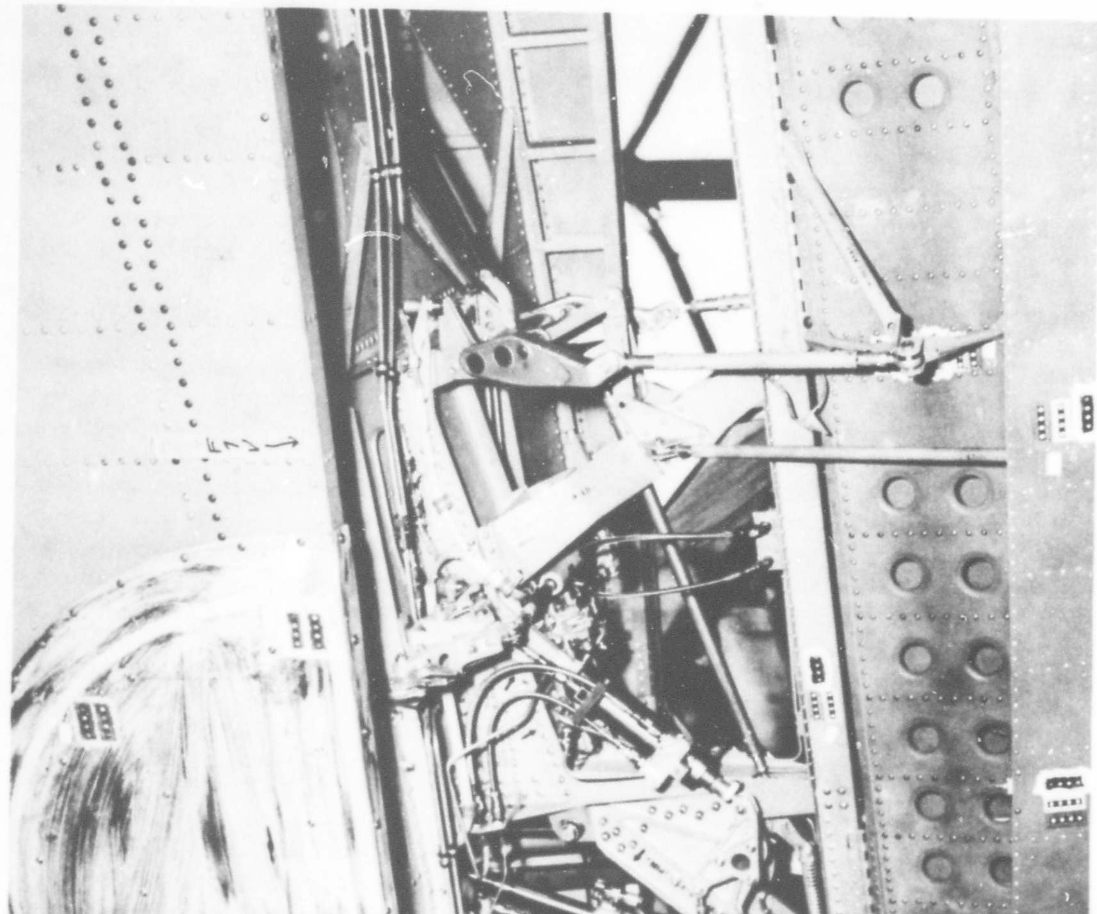


Figure 5.6 Temperature Tab Installation

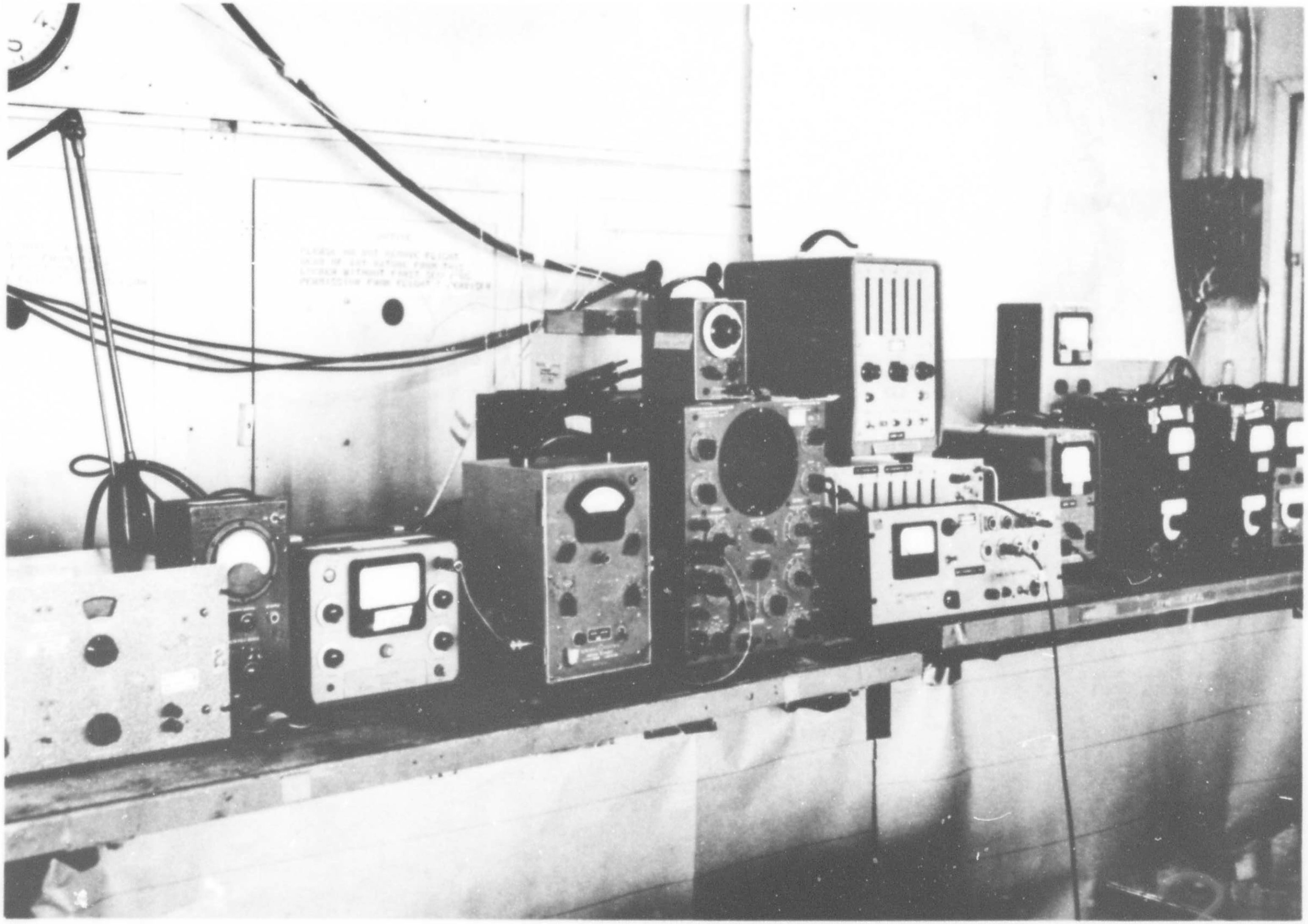


Figure 5.7 Recording Instrumentation - Engine Runs

Figure 5.8 Recording Instrumentation - Engine Runs

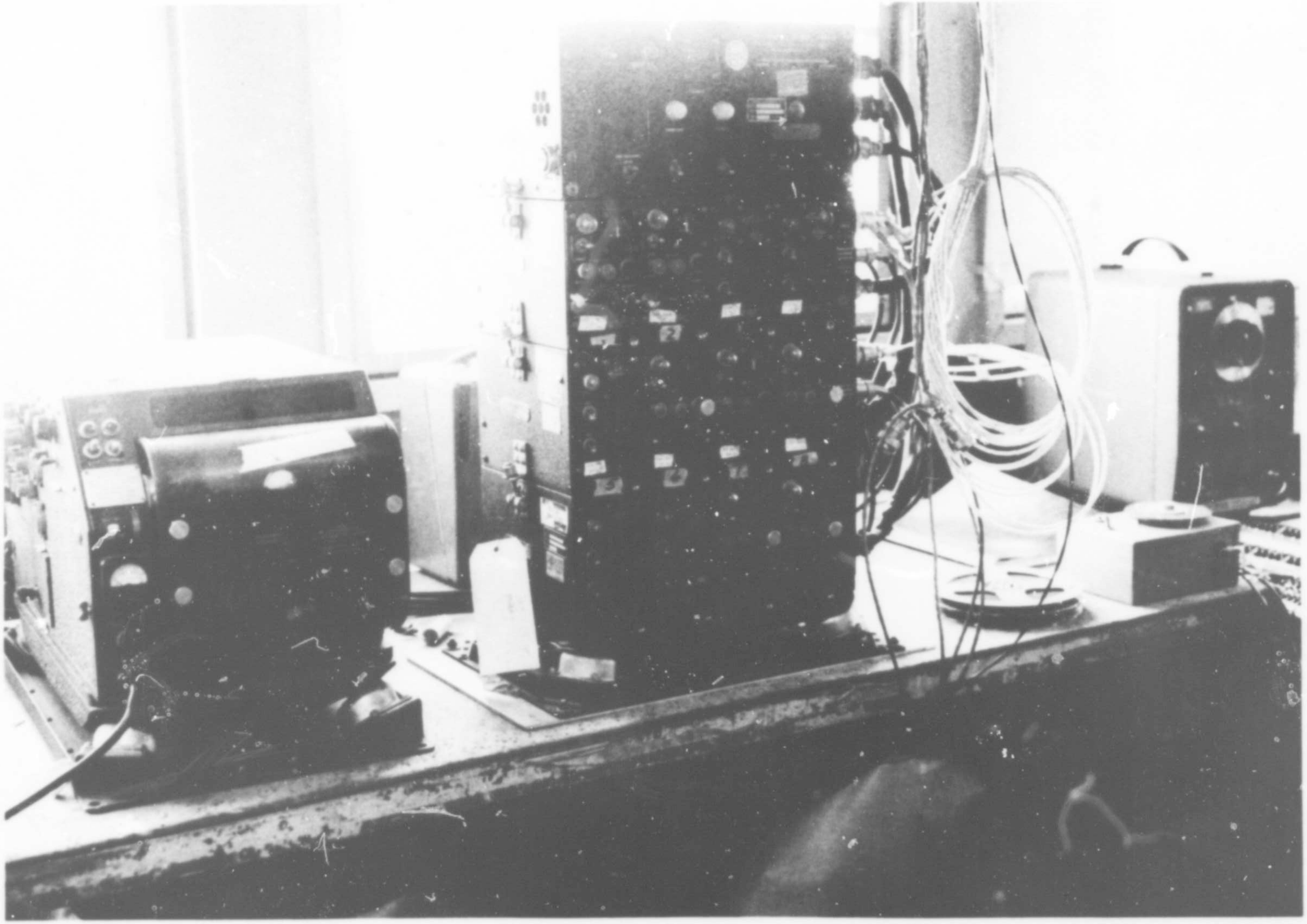


Figure 5.9 Recording Instrumentation - Engine Runs

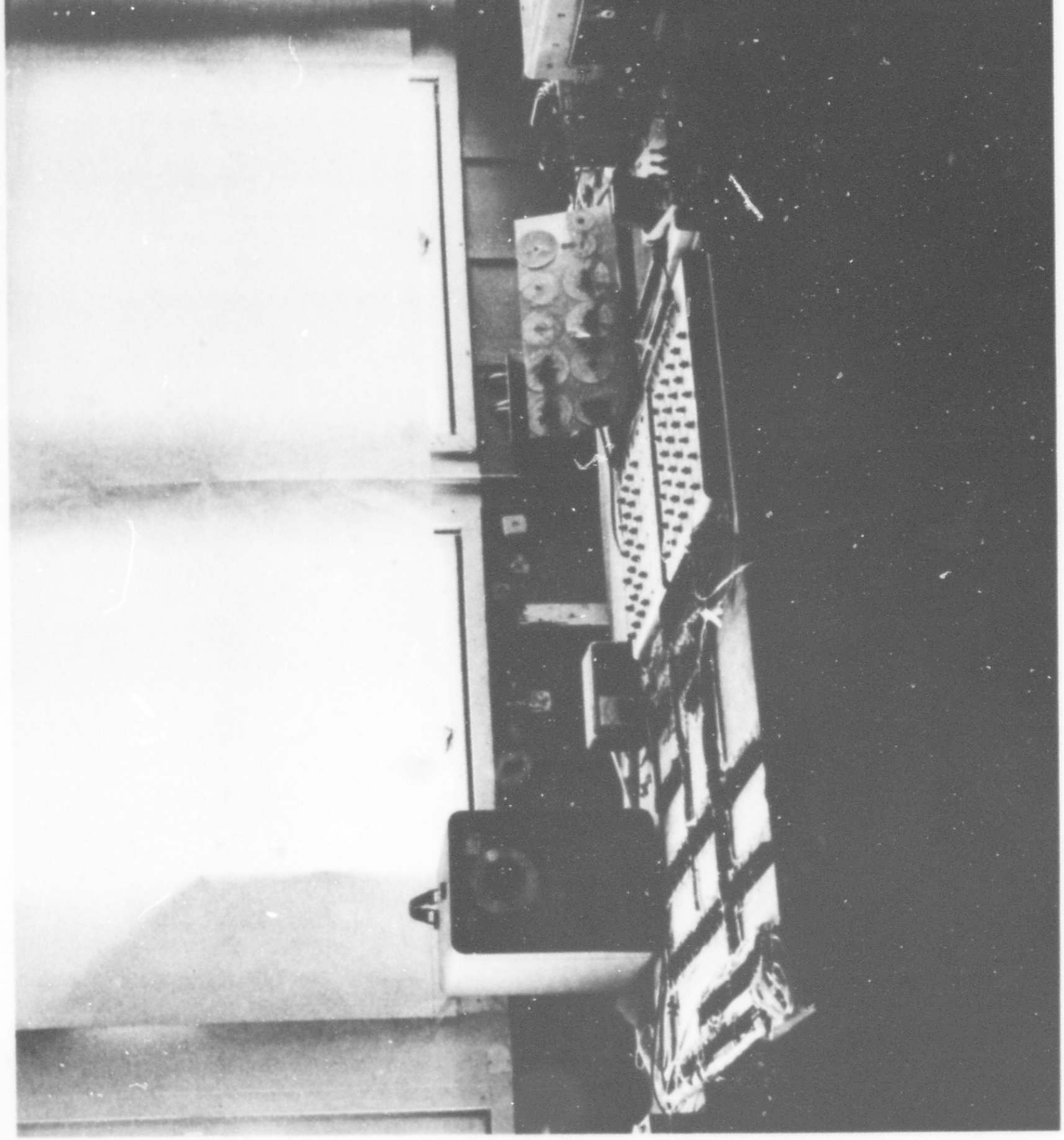


Figure 5. 10 Recording Instrumentation - Engine Runs

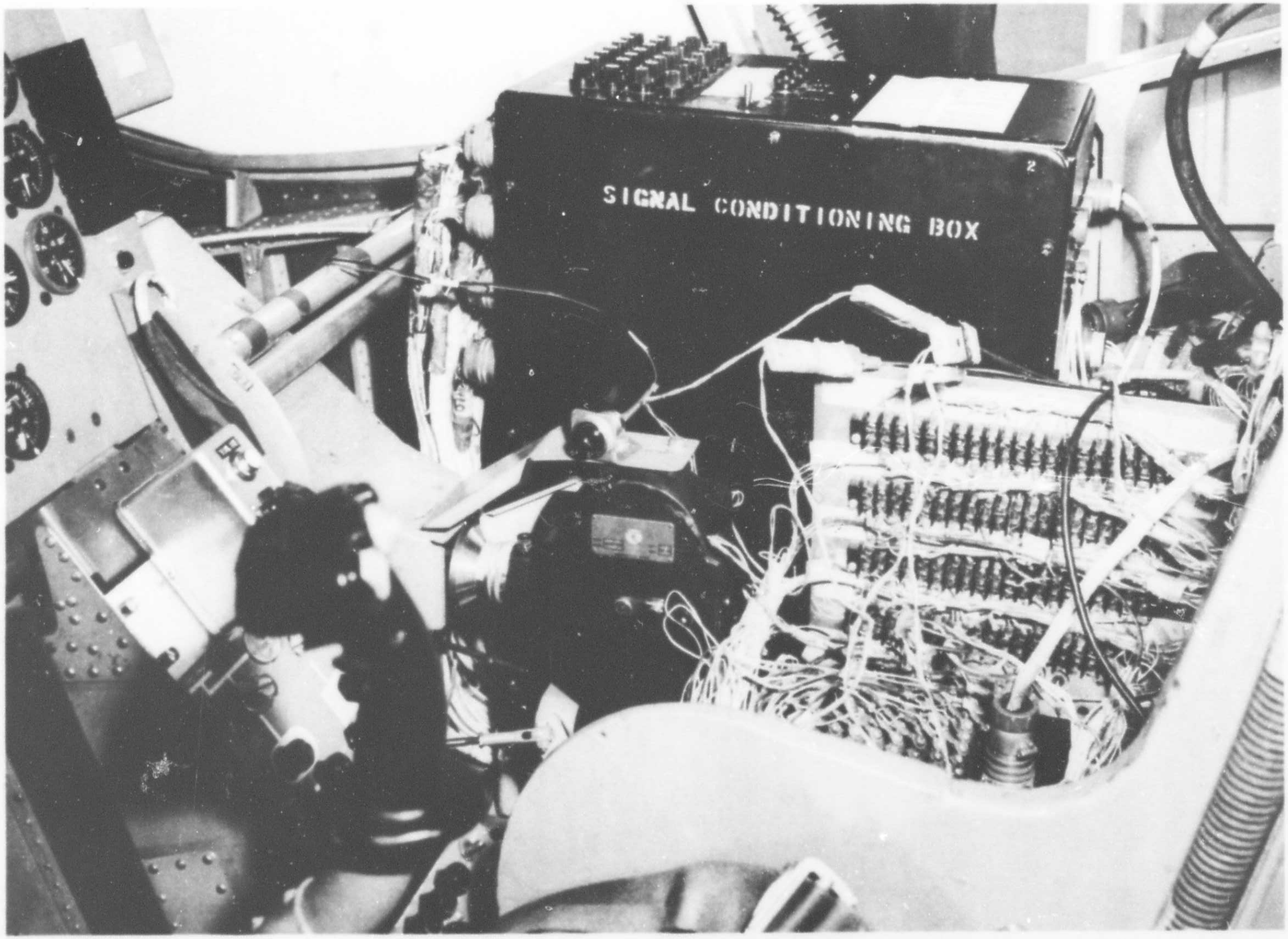


Figure 5. 11 Cockpit Instrumentation - Engine Runs

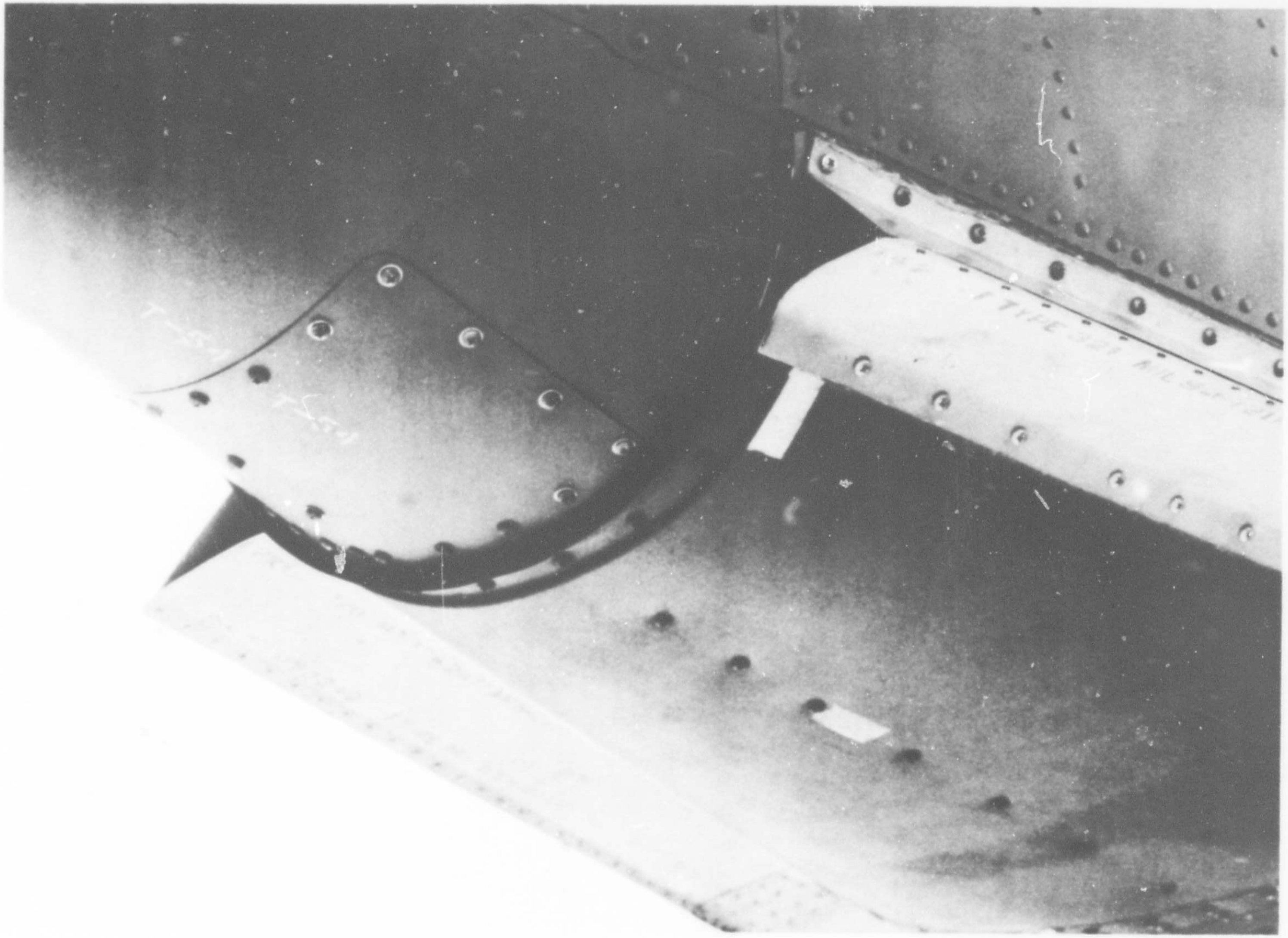


Figure 5. 12   Chine Rail Installation - Engine Run

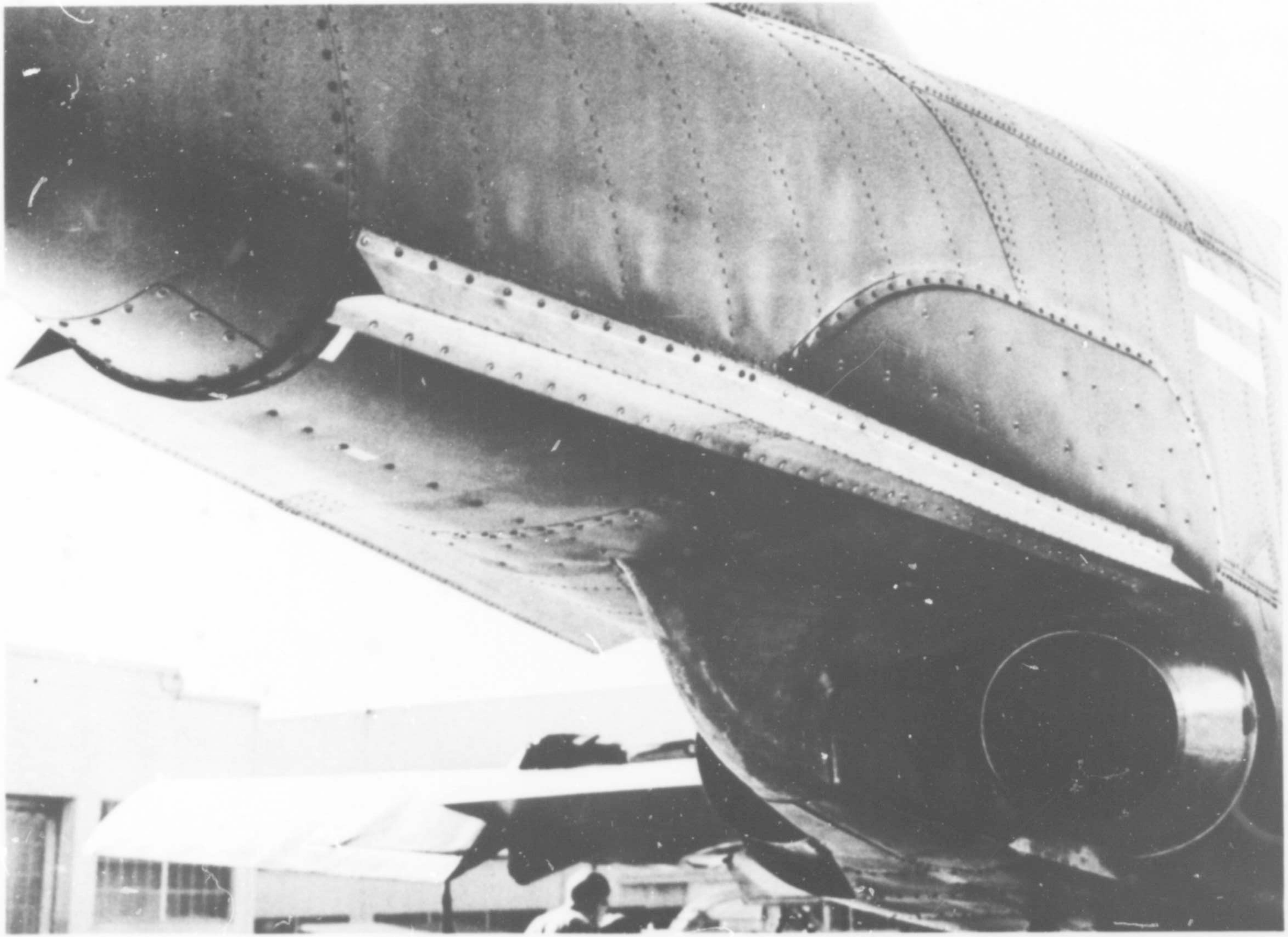


Figure 5. 13 Chine Rail Installation - Engine Run

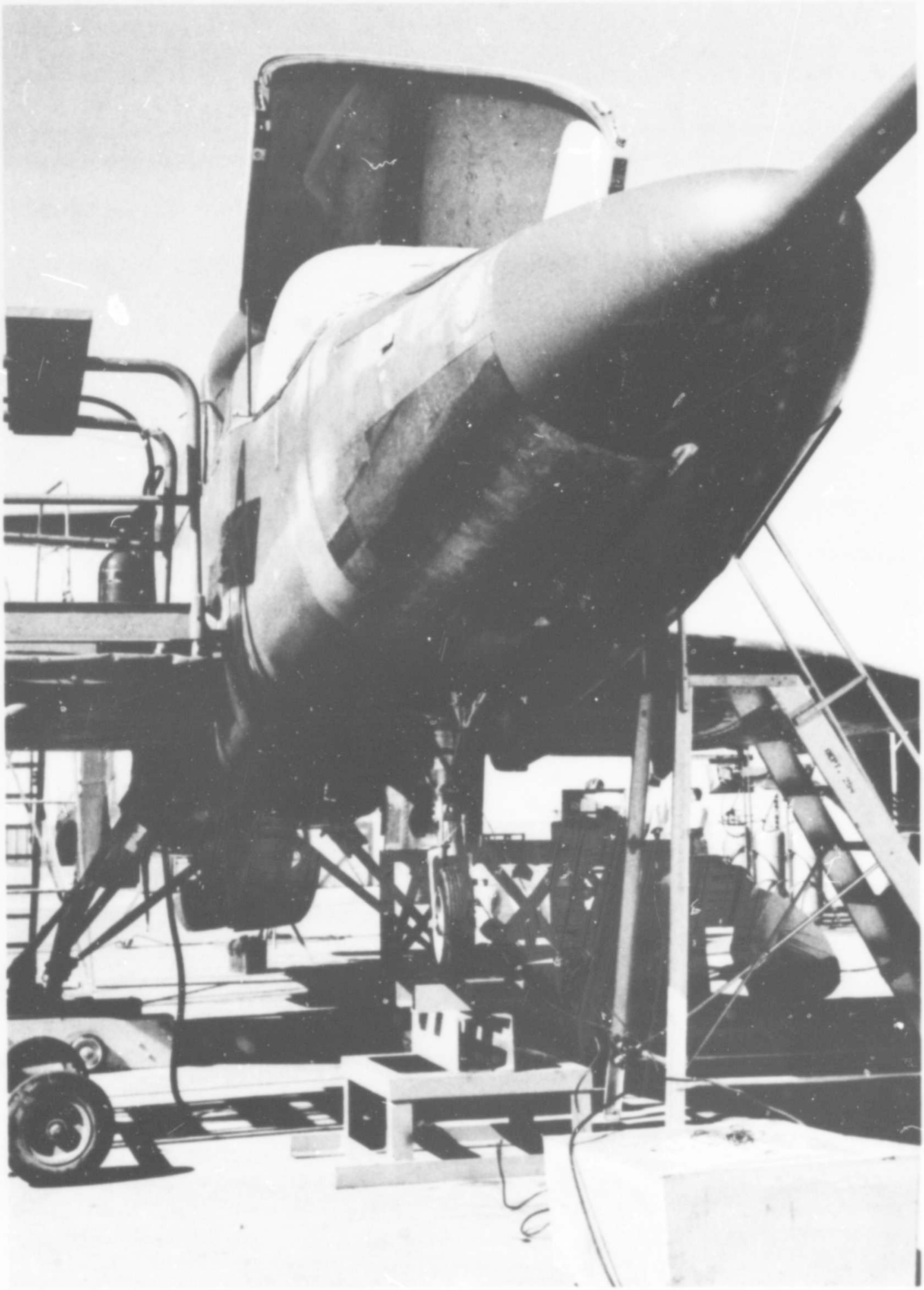


Figure 5.14 Weight - Balance Test

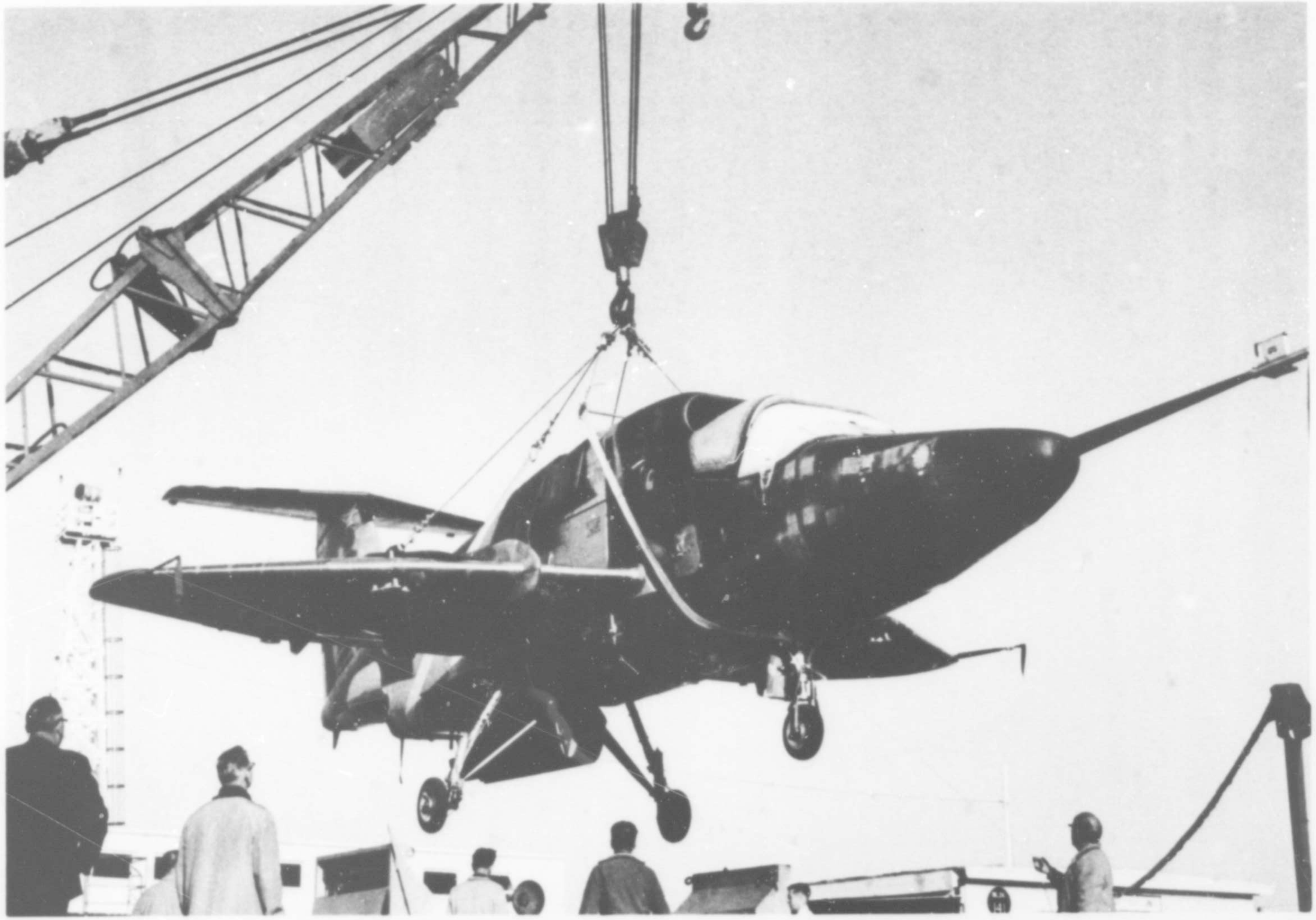


Figure 5. 15 Weight - Balance Test

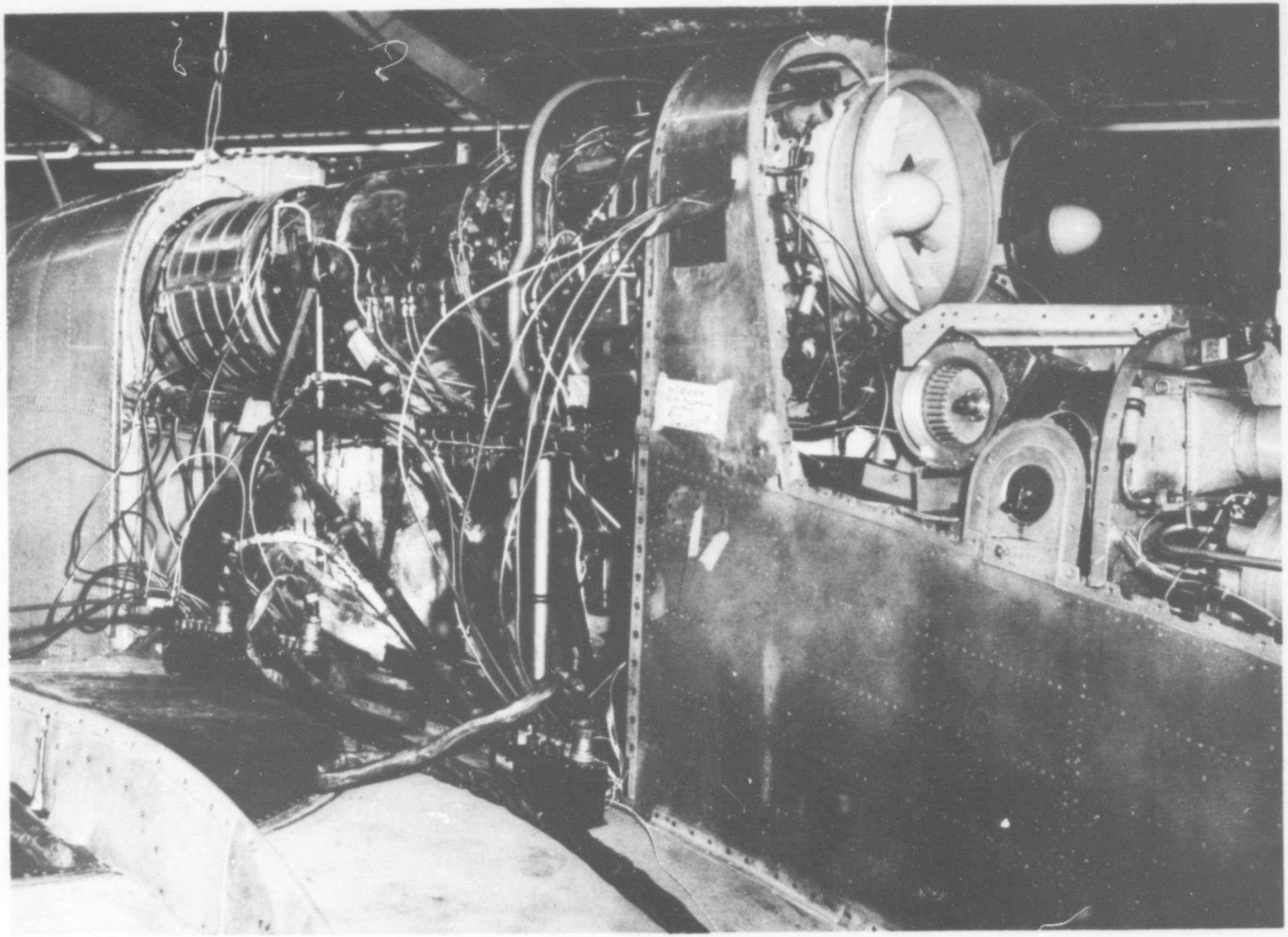


Figure 5. 16 Fire Extinguisher System Test Setup

6.0 TABLES, SCHEMATICS AND CURVES

6.1 TABLES

TABLE I

INSTRUMENTATION PARAMETER LIST

Code No.	Function	
PG-8	Cockpit	
PG-20	L/H Engine Tailpipe Ejector	
PG-24	Pitch Fan Compt Inlet Cooling Air	
PG-25	Cross Over Duct Compt	
PG-26	Cooling Fan Compt Inlet	
PG-27	Cooling Fan Inlet	
PG-28	Electronics Comp Cooling Air Inlet	
PG-29	L/H Fwd Cooling Fan Exhaust	Static Pressures
PG-30	Pitch Fan Compt Inlet Cooling Air Ejector	
PG-31	L/H Aft Cooling Blower Exhaust	
PG-35	L/H Wing Fan Aft Cooling Air Ejector	
PG-37	L/H Wing Fan Fwd Cooling Air Ejector	
PG-38	Cockpit-Rear Canopy Inlet	
PG-39	L/H Engine Comp	
PG-40	L/H Engine Bleed Duct	
TS-450	Aft Fuselage Sta 287, L/H	
TS-452	Aft Fuse. Sta 400, Lwr L/H Longeron	
TS-458	Aft Fuse. Canted Blkhd Sta 400	
TS-506	Center Fuse. - Fwd Wing Spar Lower Cap	
TS-513	Center Fuse. - Lwr Longeron Skin Flange	
TS-610	Wing R/H Rear Spar Upper Cap B. L. 44	Temperatures
TS-612	Wing [illegible]/H Rear Spar Upper Cap B. L. 25	
TS-614	Wing Fan R/H Rear Mt. Supporting Structure	
TS-618	Wing R/H Front Spar Lwr Cap B. L. 61	
TS-620	Wing R/H Panel Upr Fwd Inbd -3	

TABLE I (Continued)

INSTRUMENTATION PARAMETER LIST (Continued)

Code No.	Function	
TS-622	Wing R/H Panel Upr Aft Inbd -5	Temperatures
TS-624	Wing R/H Panel Lwr Fwd Inbd -7	
TS-701	Main Land'g Gear L/H Main Support Structure	
TS-703	Main Land'g Gear L/H Drag Strut Fold	
TS-705	Main Land'g Gear L/H "V" Brace WL 87	
TS-707	Main Land'g Gear Mode Change Cylinder @ Ring Joint	
TS-709	Main Land'g Gear L/H Axle Bet. Shock Strut and Wheel	
TS-711	Main Land'g Gear L/H Drag Brace @ Upper Pivot	
TS-725	Nose Land'g Gear Wheel Well Sta 106	
TG-13	Cooling Fan Compt. Inlet	Temperatures
TG-15	Cooling Fan Compt.	
TG-35	LH Engine Inlet Air Temp	
TG-36	R/H Engine Inlet Air Temp	
TG-5	L/H Upr Cross Over Duct Compt. Otbd.	Temperatures
TG-6	R/H Upr Cross Over Duct Compt. Otbd.	
TG-7	Pitch Fan Compt. Cooling Air Inlet	
TG-11	Cross Duct Compt.	
TG-16	Pitch Fan Comp. Cooling Air Ejector Inlet	
TG-17	L/H Fwd Cooling Fan Exhaust	
TG-19	L/H Aft Cooling Fan Exhaust	
TG-21	L/H Engine Compressor Section Exhaust	
TG-27	L/H Wing Fan Aft Cooling Air Ejector	

TABLE I (Continued)

INSTRUMENTATION PARAMETER LIST (Continued)

Code No.	Function	
TG-29	L/H Wing Fan Fwd Cooling Air Ejector	
TS-623	L/H Wing Panel Lwr Fwd Inbd -7	
TS-625	L/H Wing Panel Lwr Aft Inbd -9	
TS-262	R/H Wing Panel Lwr Aft Inbd -9	
TS-641	Wing Fan Bearing - Upper L/H	
TS-642	Wing Fan Bearing - Upper R/H	Temperatures
TS-643	Wing Fan Bearing - Lower L/H	
TS-644	Wing Fan Bearing - Lower R/H	
TS-305	Pitch Fan Bearing	
TS-801	Fwd Ldg Gear Door Idler Link - L/H	
TS-802	Fwd Ldg Gear Door Rod (Steel)	
TS-803	Aft Ldg Gear Door 031-37 Rod	
TS-805	Aft Ldg Gear Support Rod - L/H	
TS-807	L/H Aft Access Panel	
TS-301	Pitch Fan Side Mt. L/H Support Struc	
TS-304	Pitch Fan Front Frame	
TS-508	Center Fuse MLG Door Sill @ Flap Support L/H	
TS-455	Aft Fuse Lwr Long. Flange @ Canted Fr. 43T005	
TS-601	Wing L/H Rear Spar Lwr Cap BL 26	
TS-603	Wing L/H Rear Spar Lwr Cap BL 44	Temperatures
TS-605	Wing L/H Rear Spar Lwr Cap BL 61	
TS-607	Wing L/H Rear Spar Lwr Cap BL 71	
TS-609	Wing L/H Rear Spar Upr Cap BL 44	
TS-611	Wing L/H Rear Spar Upr Cap BL 25	
TS-613	Wing Fan L/H Rear Mt. Support'g Struc	

TABLE I (Continued)

INSTRUMENTATION PARAMETER LIST (Continued)

Code No.	Function	
TS-615	Wing L/H Front Spar Lwr Cap	Temperatures
TS-617	Wing L/H Front Spar Lwr Cap	
TS-619	Wing L/H Upr Fwd Inbd -3	
TS-621	Wing L/H Panel Upr Aft Inbd -5	
TS-627	Wing L/H Fairing Inbd BL 25	
TS-629	Wing L/H Fairing Inbd BL 56	
TS-630	Wing L/H - 143X025 Brkt 47 Lwr Flange	
TS-661	Flap L/H 143#010-101	
TS-602	Wing R/H Rear Spar Lwr Cap BL 26	
TS-604	Wing R/H Rear Spar Lwr Cap BL 44	
TS-606	Wing R/H Rear Spar Lwr Cap BL 61	
TS-608	Wing R/H Rear Spar Lwr Cap BL 71	
TL-1	L/H Engine Oil into Tank Temp	Temperatures
TL-2	R/H Engine Oil into Tank Temp	
TG-25	L/H Engine Tailpipe Ejector	
TG-32	L/H Wing Fan Inlet Air Temp	
TG-23	L/H Engine Turbine Section	
TS-713	MLG Door L/H-3 Inner Panel -123 Sk Flange	
TS-715	MLG Door L/H -3 Inner Panel -123 Sk Flange	
TS-717	MLG Door L/H -1 Inner Panel -4 Sk Flange	
TS-719	MLG Door L/H -1 Inner Panel -9 Sk Flange	
TS-302	Pitch Fan Aft Mt Spt Structure	
TS-303	Pitch Fan Aft Hinge Frame	
TS-651	Flap L/H Inbd Fitting	

TABLE I (Continued)

INSTRUMENTATION PARAMETER LIST (Continued)

Code No.	Function	
TS-501	Space Frame -59, Mbr 3 to 8 Sta	Temperatures
TS-502	Space Frame -41, Mbr 4 to 20 Upr Diag Sta 251	
TS-503	Space Frame -73, Mbr 8 to 13 Upr Long Sta 272	
TS-504	Space Frame -79 Mbr 8 to 25 Side Diag Sta 257	
TS-505	Space Frame -249 Mbr 25 to 28 Lwr Long Sta 245	
TS-514	Space Frame -46 & -48 & -18B Jct.	
TS-507	Center Fuse - Lwr Access Fairing STS 256 Upr	
TS-453	Aft Fuse Exh Shroud Sta 400, WL 94, BL 21	
TS-454	Aft Fuse Exh. Shroud L/H	
TS-457	Aft Fuse Vert Stab Frt Spar Frame 143T005	
TG-31	R/H Wing Fan Inlet Air Temp	
TG-8	Cockpit Compt Temp	Temperatures
TG-4	Electronic Compt Cooling Air Inlet	
TS-721	MLG L/H Shock Strut - Top of Oleo	
TL-3	Hydraulic Res Fluid Temp L/H	
TL-4	Hydraulic Res Fluid Temp R/H	
TS-306	Pitch Fan Frt. Frame (T/C On Casting I. D.)	
TS-307	Pitch Fan Frt. Frame (T/C On Casting I. D.)	
TS-308	Pitch Fan Frt. Frame (T/C On Strut Near Hub)	
TS-309	Pitch Fan Frt. Frame	
TS-509	L/H Generator	
TS-510	R/H Generator	
TS-511	Forward Inverter	
TS-512	Aft Inverter	
TS-460	L/H Diverter Valve Actuator Temp	
TS-461	R/H Diverter Valve Actuator Temp	

TABLE I (Continued)

INSTRUMENTATION PARAMETER LIST (Continued)

Code No.	Function	
TG-2	Exhaust Gas Temp - L/H Engine	
TG-3	Exhaust Gas Temp - R/H Engine	
TS-462	L/H Engine Lwr Turbine Section Flange	
TG-2	Exhaust Gas Temp - L/H Engine	
TG-3	Exhaust Gas Temp - R/H Engine	
TS-463	R/H Engine Lwr Turbine Section Flange	
TG-2	Exhaust Gas Temp - L/H Engine	Temperatures
TG-3	Exhaust Gas Temp - R/H Engine	
TS-451	Aft Fuse Exh Duct Sta 400, WL 94 R/H	
TG-2	Exhaust Gas Temp - L/H Engine	
TG-3	Exhaust Gas Temp - R/H Engine	
TS-456	Aft Fuse Exh. Duct, Sta 400	
TG-2	Exhaust Gas Temp - L/H Engine	
TG-3	Exhaust Gas Temp - R/H Engine	
TG-33	L/H Cross Over Duct - Inside Shroud	
TG-2	Exhaust Gas Temp - L/H Engine	
TG-3	Exhaust Gas Temp - R/H Engine	
TG-34	R/H Cross Over Duct - Inside Shroud	Temperatures
TG-2	Exhaust Gas Temp - L/H Engine	
TG-3	Exhaust Gas Temp - R/H Engine	
	Spare	
TG-2	Exhaust Gas Temp - L/H Engine	
TG-3	Exhaust Gas Temp - R/H Engine	
PO-6	L/H Aileron Position	
PO-7	R/H Aileron Position	
PO-8	Flap Position	
PO-9	Horizontal Stabilizer Position	
PO-13	Wing Fan Louver Servo Position L/H Odd	
PO-14	Wing Fan Louver Servo Position L/H Even	

TABLE I (Continued)

INSTRUMENTATION PARAMETER LIST (Continued)

Code No.	Function
PO-15	Wing Fan Louver Servo Position R/H Odd
PO-16	Wing Fan Louver Servo Position R/H Even
PO-18	Wing Fan Inlet Door Position L/H Inbd
PO-19	Wing Fan Inlet Door Position L/H Outbd
PO-20	Wing Fan Inlet Door Position R/H Inbd
PO-21	Wing Fan Inlet Door Position R/H Outbd
PO-25	L/H Diverter Valve
RPM 1	LH Engine
RPM 2	RH Engine
RPM 3	LH Fan
RPM 4	RH Fan
RPM 5	Pitch Fan

TABLE II

UNUSABLE FUEL VS. AIRCRAFT ATTITUDE

Attitude	Tank	Unusable Fuel
Level		
	Forward	5 pounds
	Aft	6
5° (Nose Up)		
	Forward	4
	Aft	4
8° (Nose Up)		
	Forward	4
	Aft	22
15.7° (Nose Up)		
	Forward	56
	Aft	67
-5.2° (Nose Down)		
	Forward	3
	Aft	5
-10.5° (Nose Down)		
	Forward	5
	Aft	24.5
-15° (Nose Down)		
	Forward	7.5
	Aft	45

6.2 SCHEMATICS

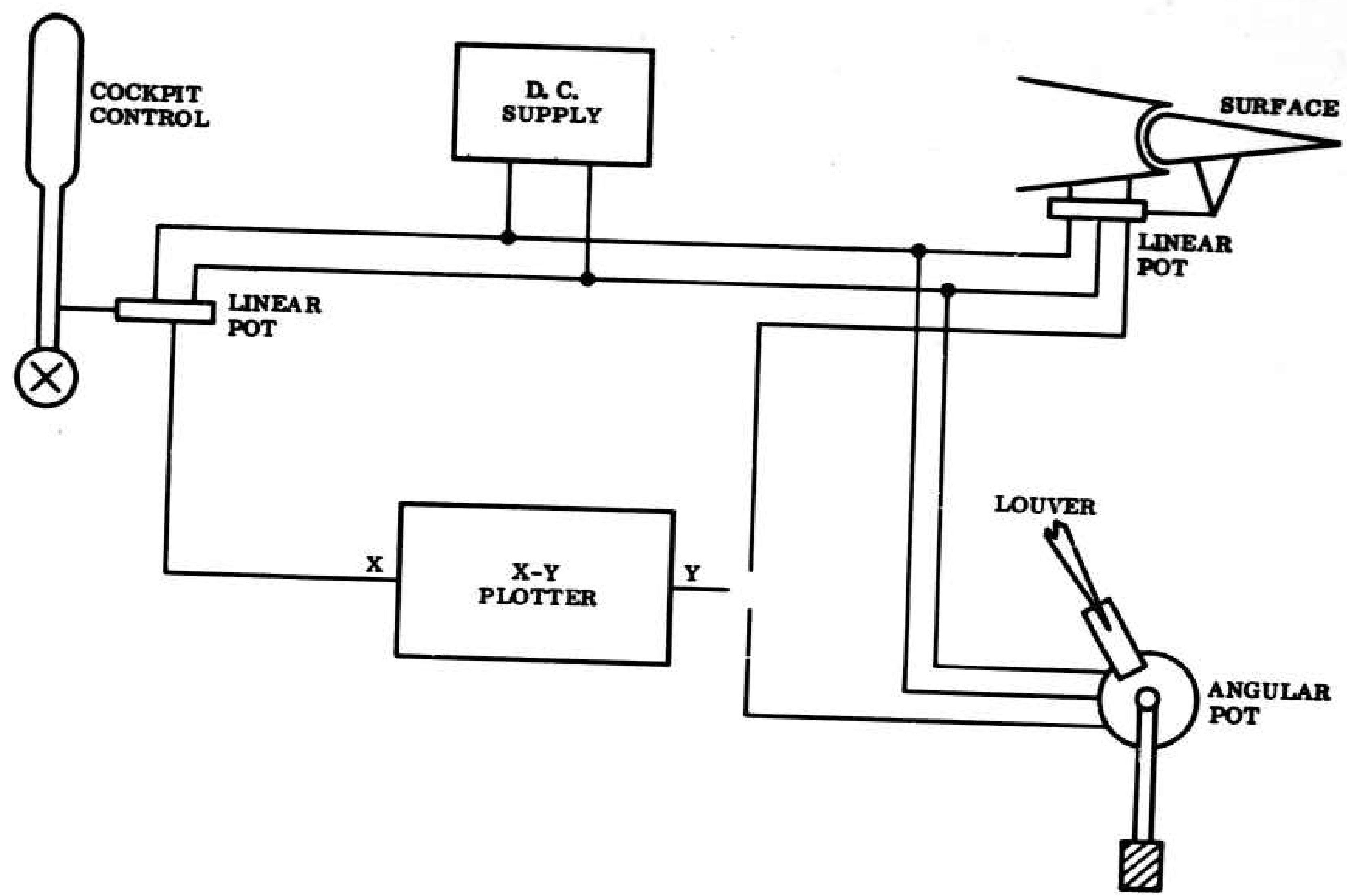


Figure 6. 2. 1 Surface Gains and Backlash Test Schematic (Surface Gains and Hysteresis)

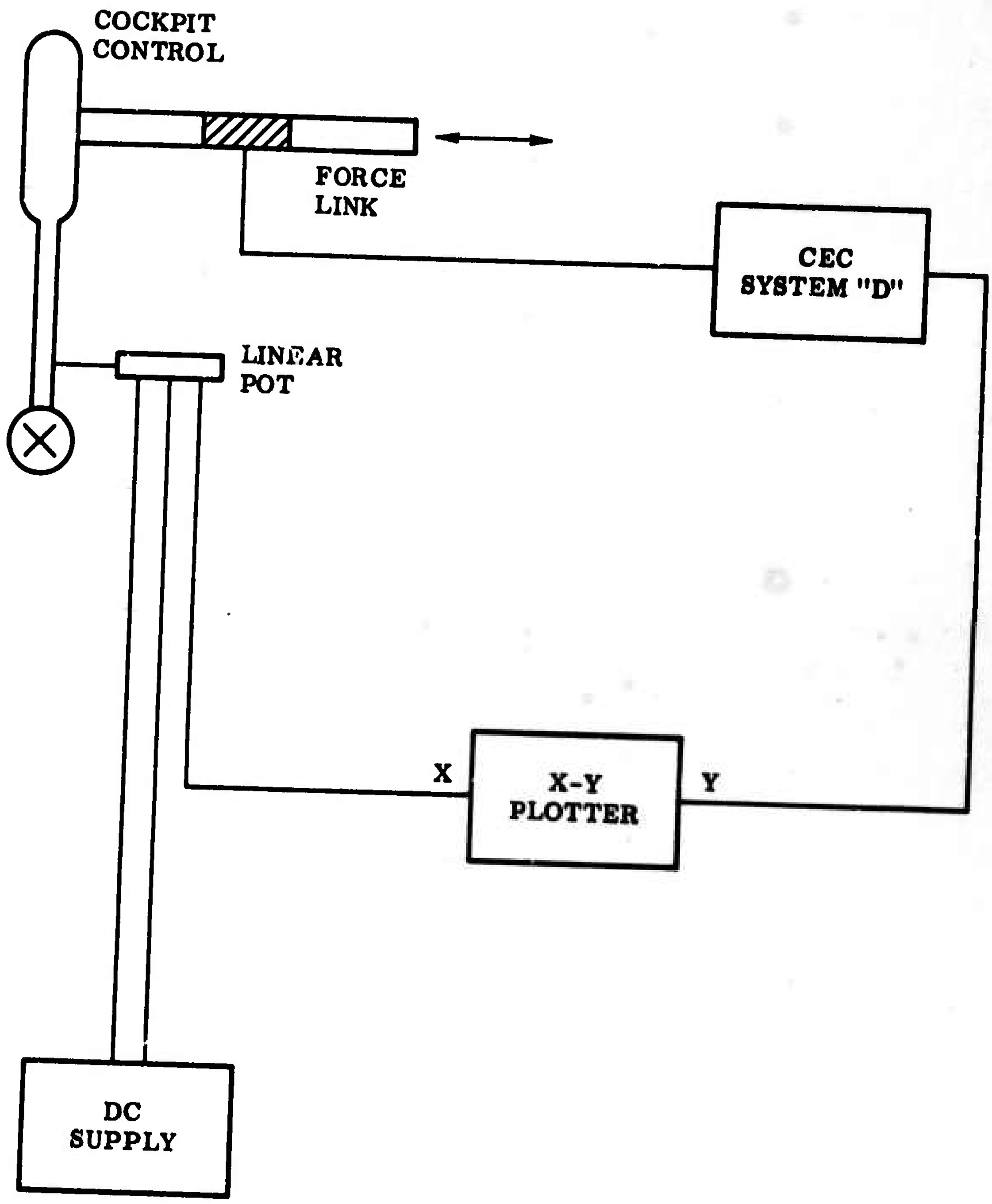


Figure 6.2.2 Control Force Test Schematic (Surface Gains and Hysteresis)

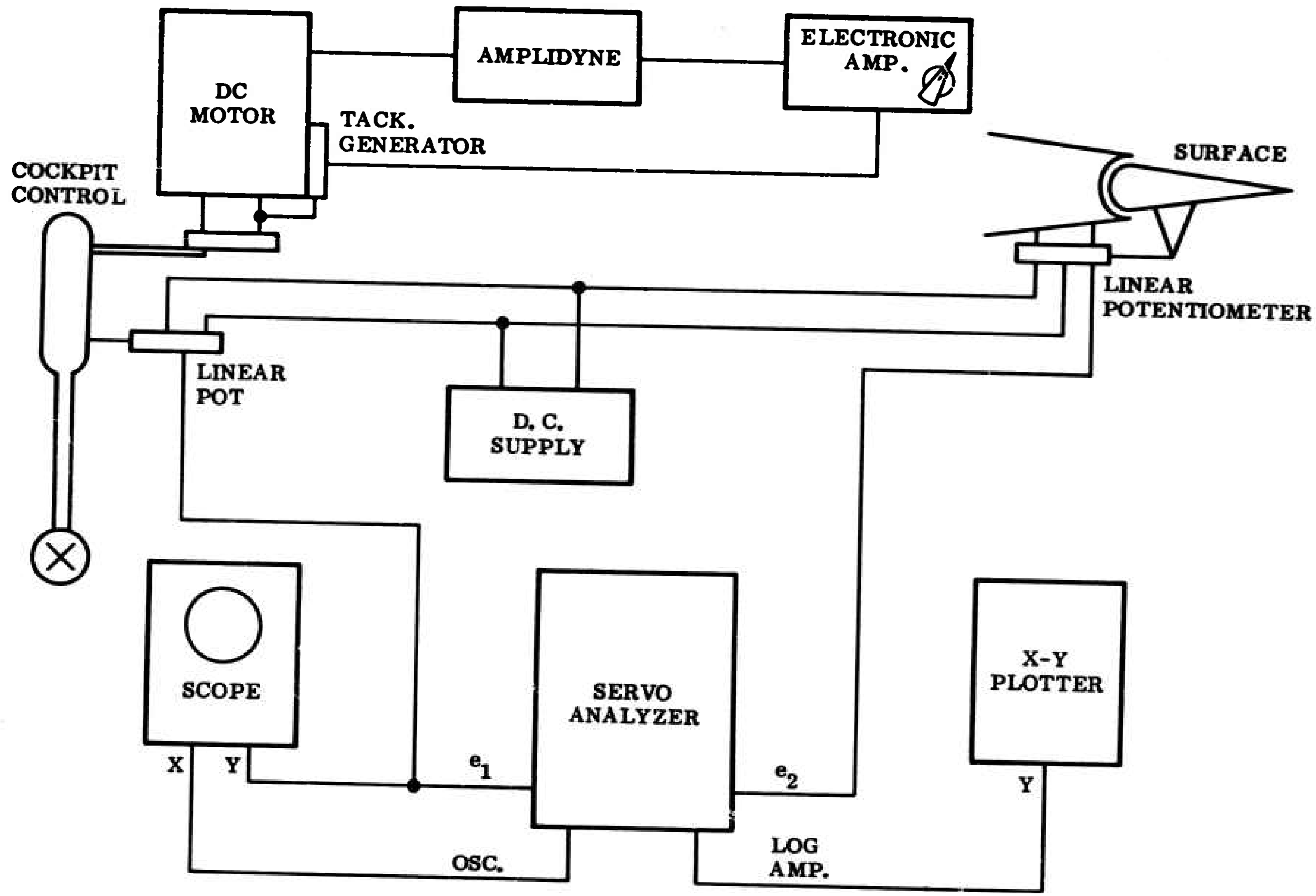


Figure 6.2.3 Frequency Response Test Schematic (Flight Controls Stability)

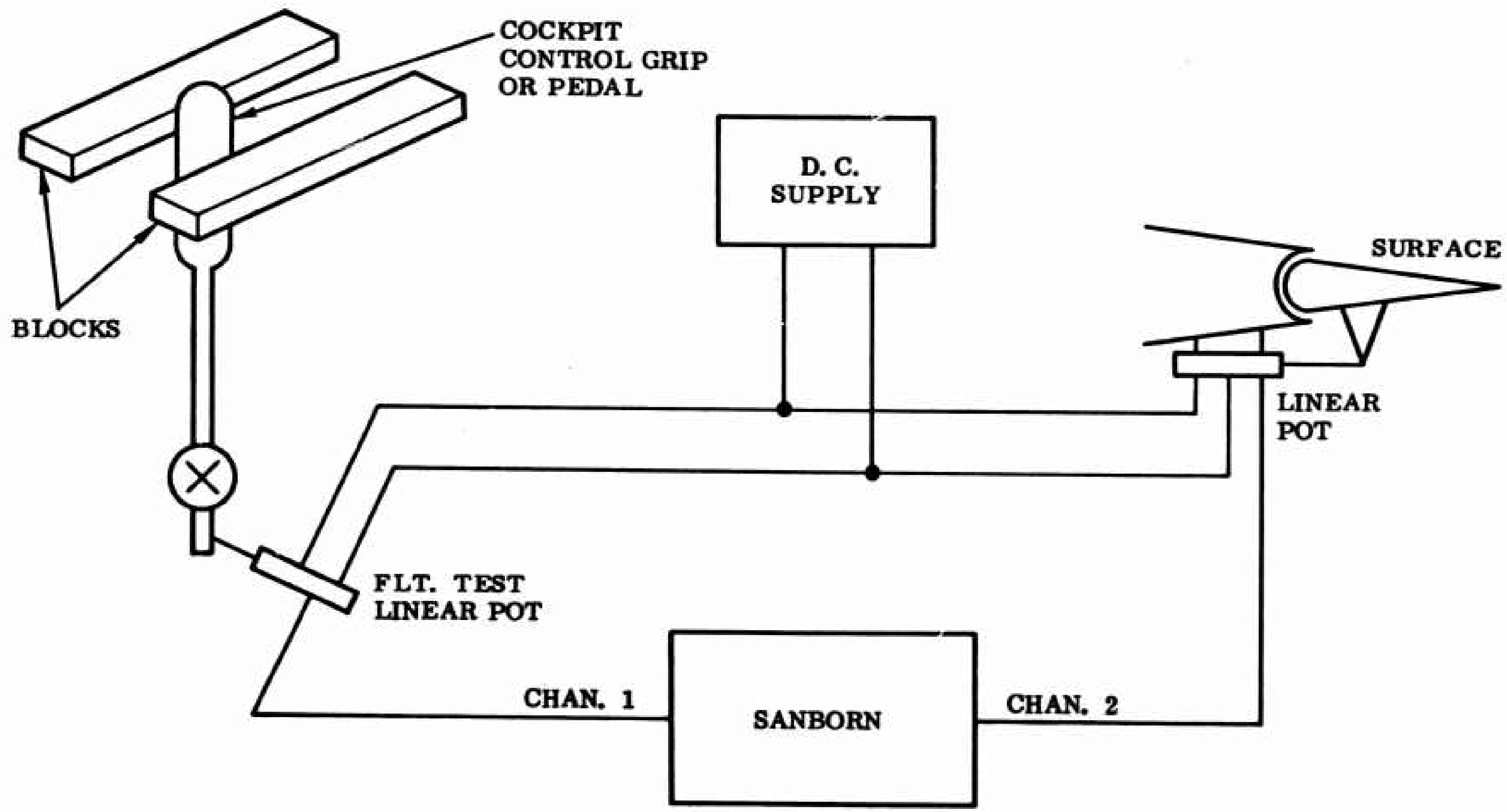


Figure 6. 2. 4 Transient Response Test Schematic (Flight Controls Stability)

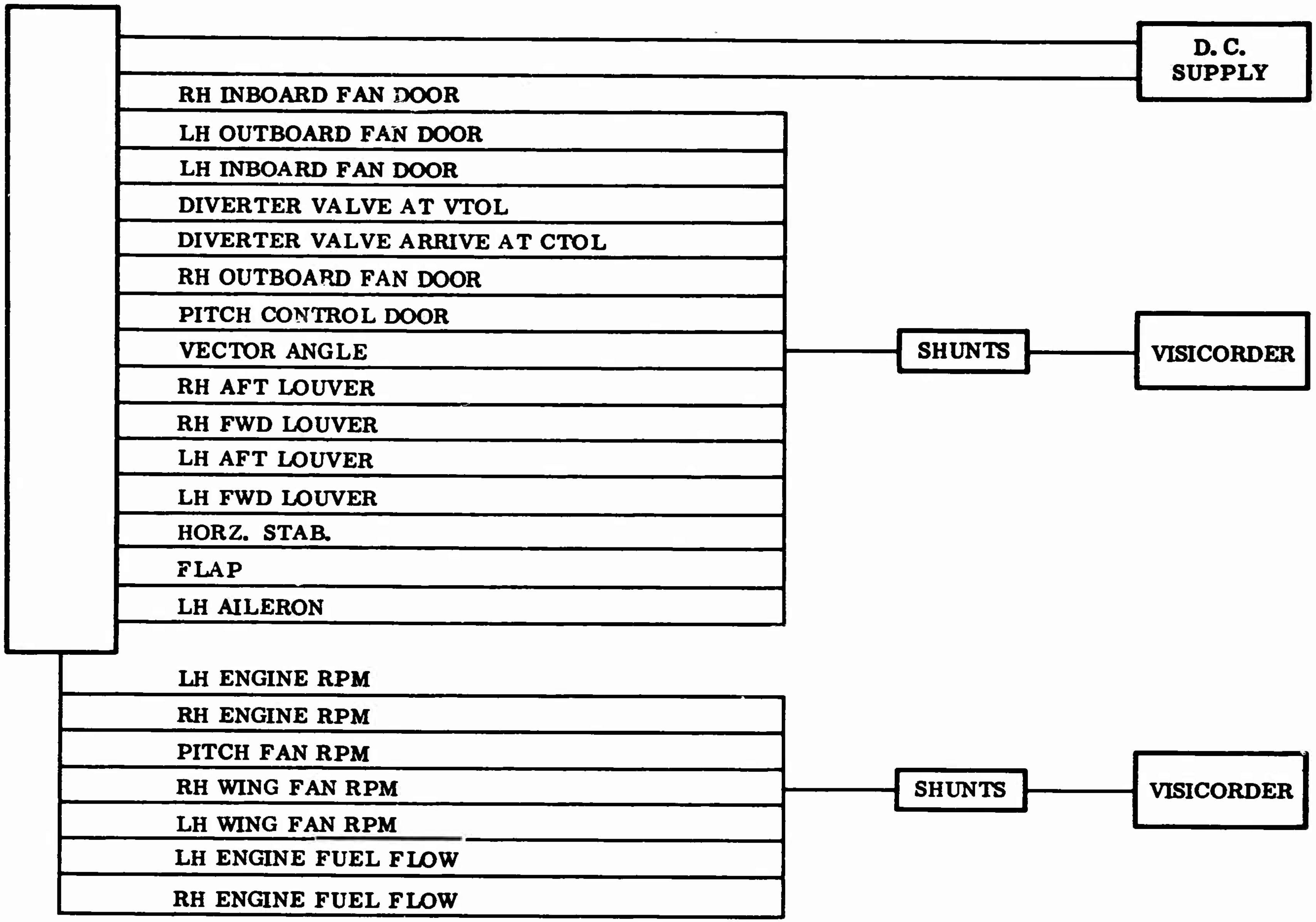


Figure 6. 2. 5 Flight Mode Conversion Sequence and Fan Flight Trim Rates Parameter Outline

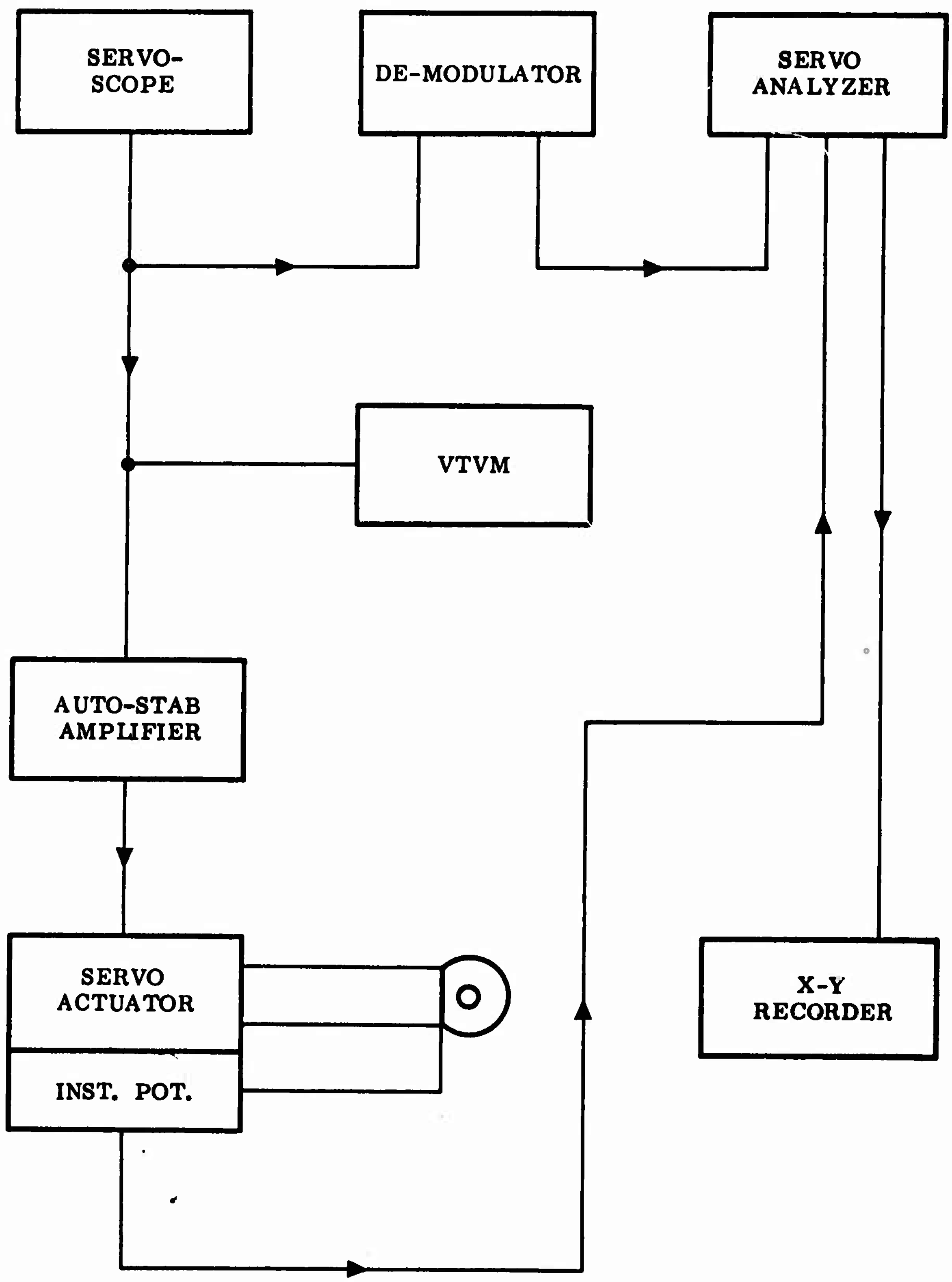


Figure 6.2.6 Auto-Stability Frequency Response Test Schematic

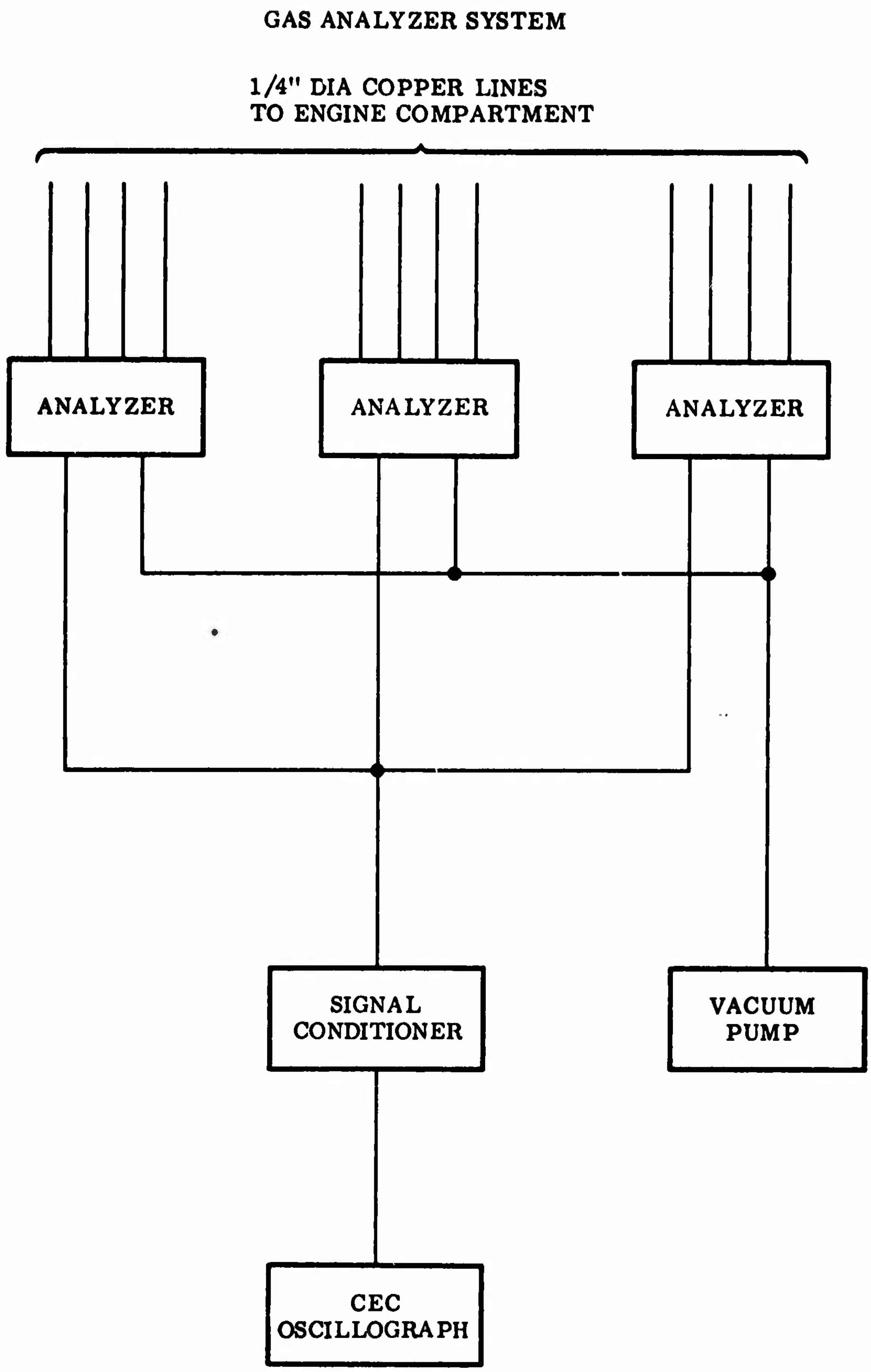


Figure 6.2.7 Fire Extinguisher System Test Schematic

6.3 CURVES

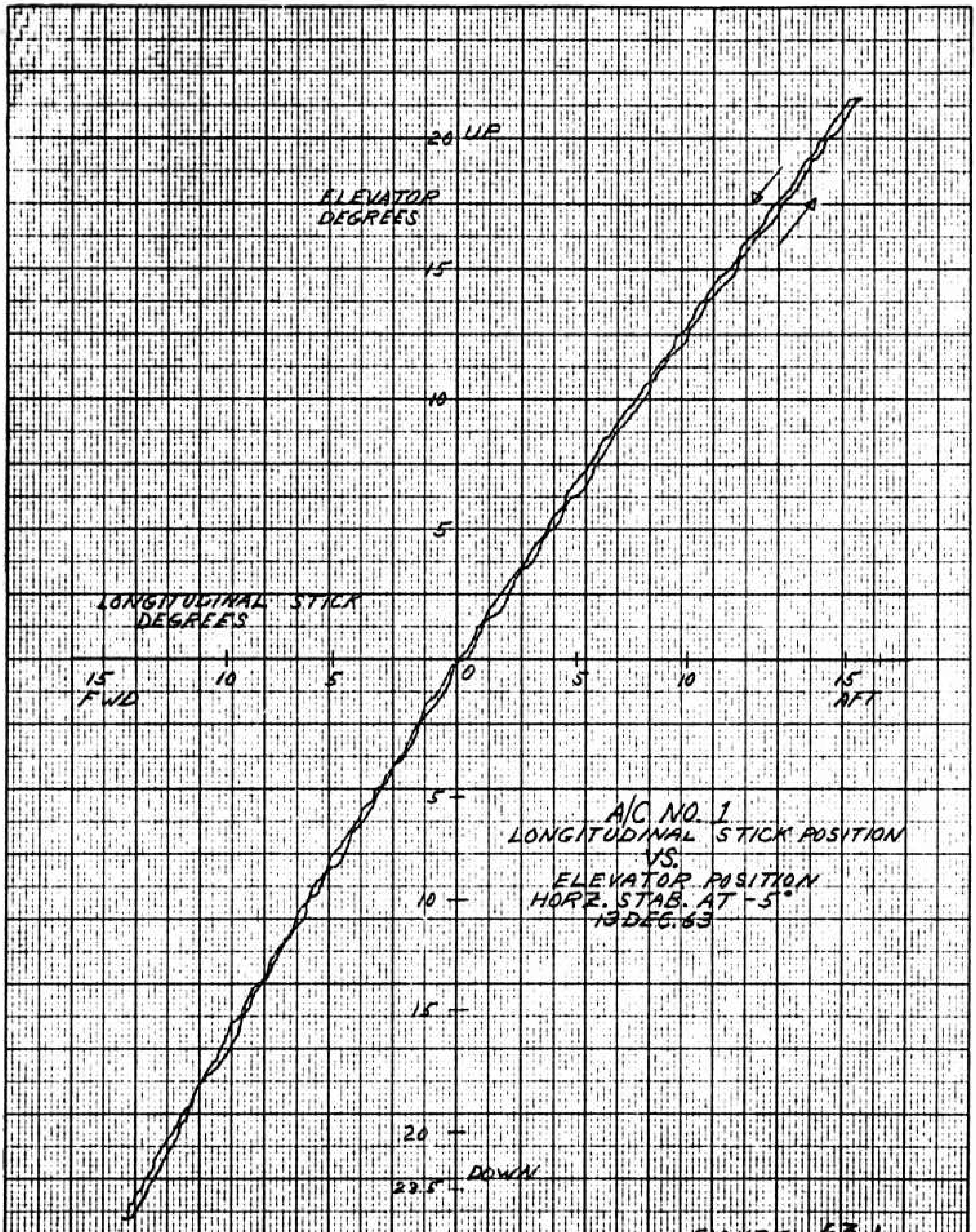
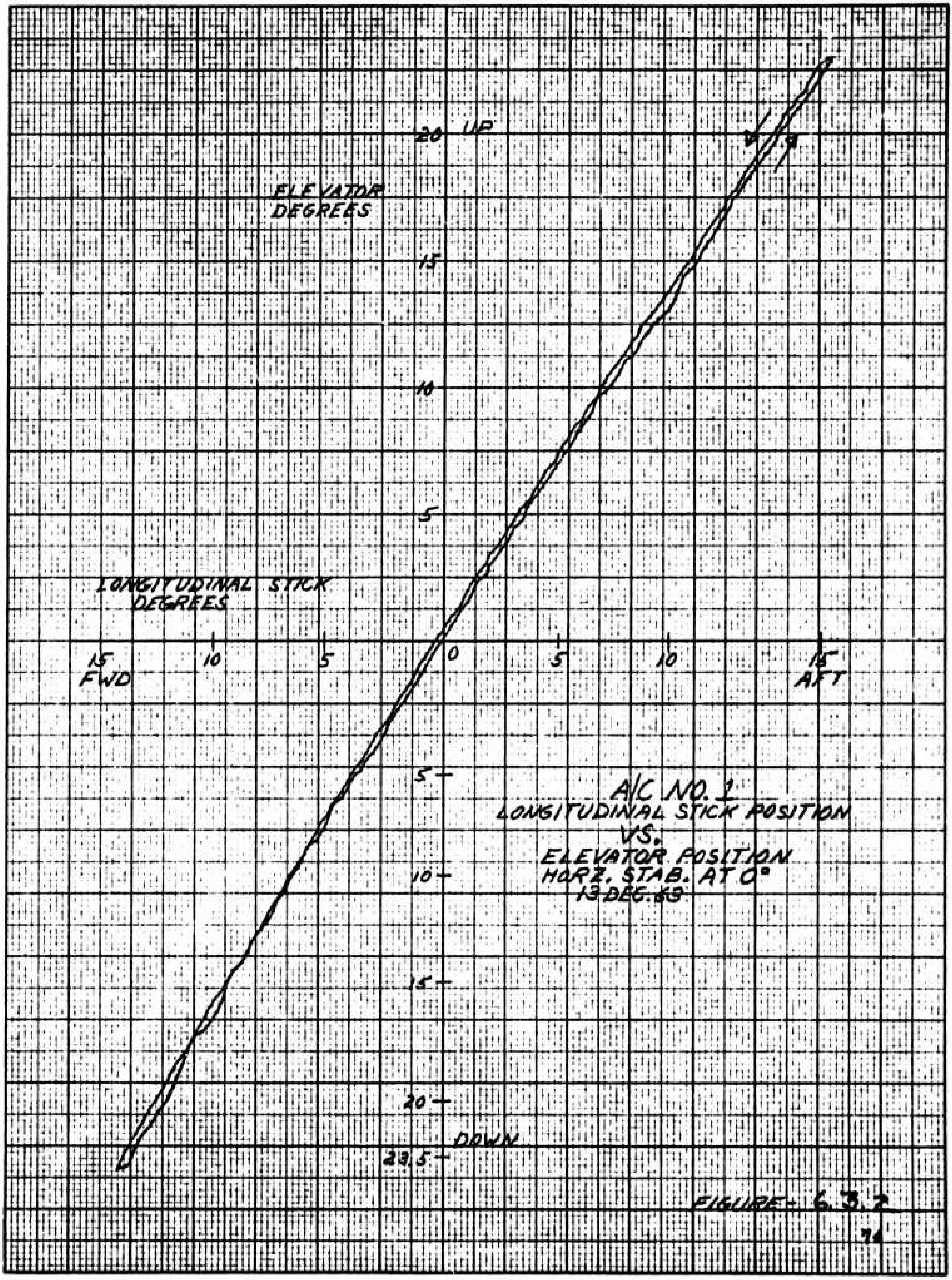
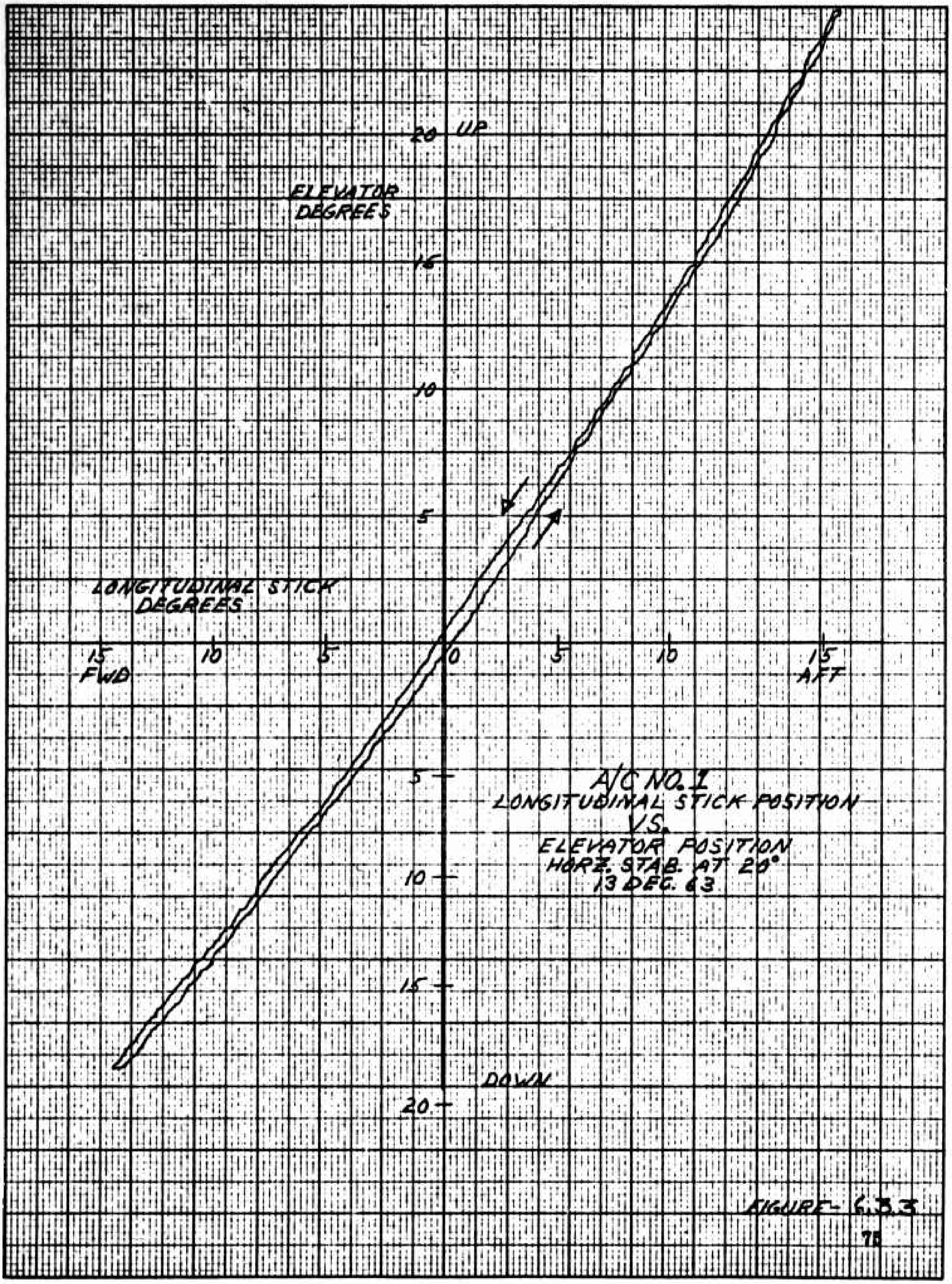

20 UP
ELEVATOR DEGREES
15
10
5
LONGITUDINAL STICK DEGREES
15 FWD
10
5
0
5
10
15 AFT
5
10
15
20
23.5
DOWN
A/C NO. 1
LONGITUDINAL STICK POSITION
VS.
ELEVATOR POSITION
HORZ. STAB. AT -5°
13 DEC. 63

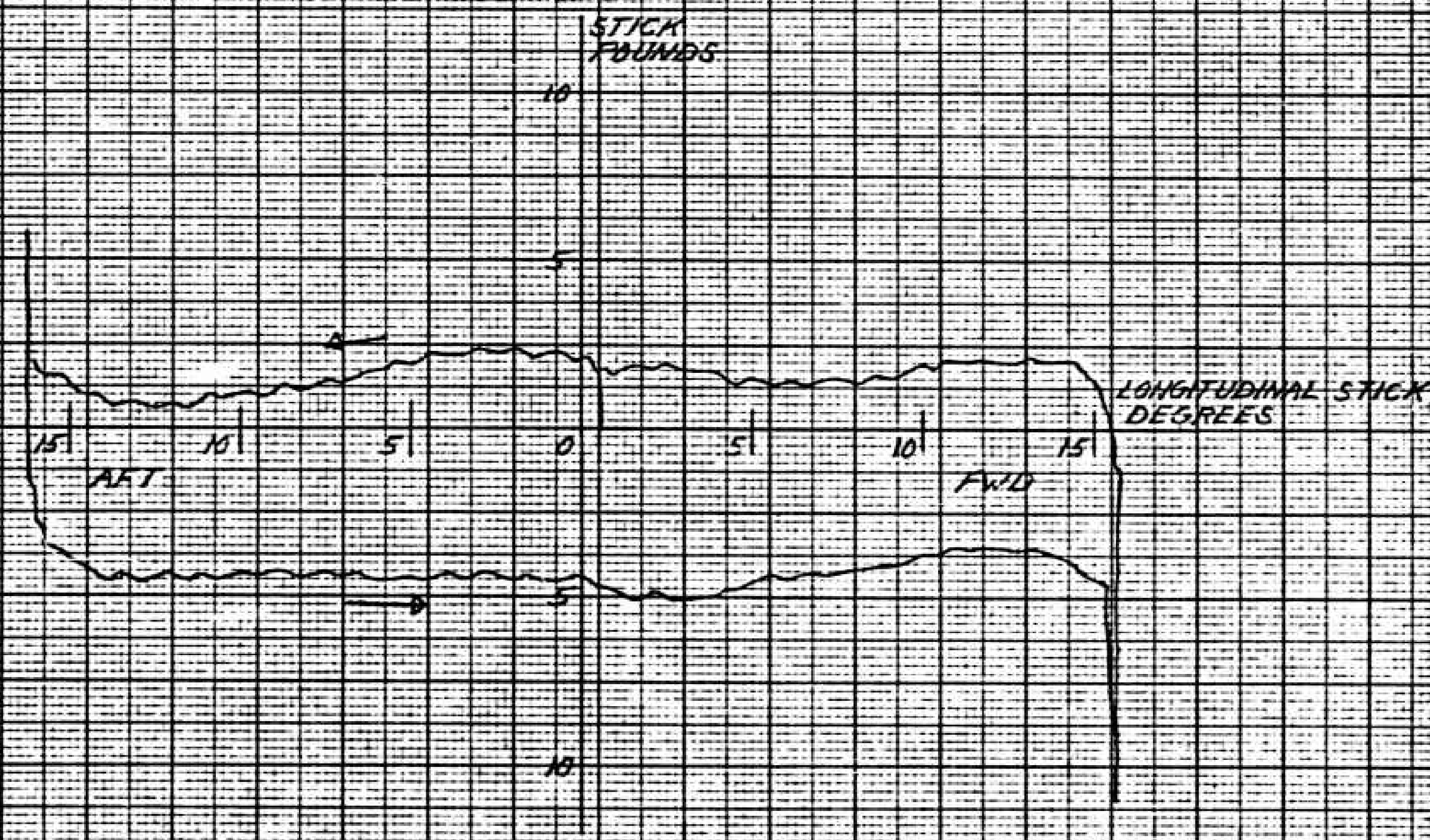
FIGURE- 63.1


ELEVATOR DEGREES
20 UP
15
10
5
LONGITUDINAL STICK DEGREES
15 FWD
10
5
0
5
10
15 AFT
5
10
15
20
23.5
DOWN
A/C NO. 1
LONGITUDINAL STICK POSITION
VS.
ELEVATOR POSITION
HORZ. STAB. AT 0°
13 DEC. 63
FIGURE 6.3.7



20 UP
ELEVATOR DEGREES
15
10
5
LONGITUDINAL STICK DEGREES
15 FWD
10
5
0
5
10
15 AFT
5
10
15
20
DOWN
A/C NO. 1
LONGITUDINAL STICK POSITION
VS.
ELEVATOR POSITION
HORZ. STAB. AT 20°
13 DEC. 63

FIGURE - 6.3.3



NO. 1 A/C
LONGITUDINAL STICK POSITION
VS.
LONGITUDINAL STICK FORCE
CTOL MODE
10 FEB. 64

FIGURE 6.3.4

FIGURE-6.3.4

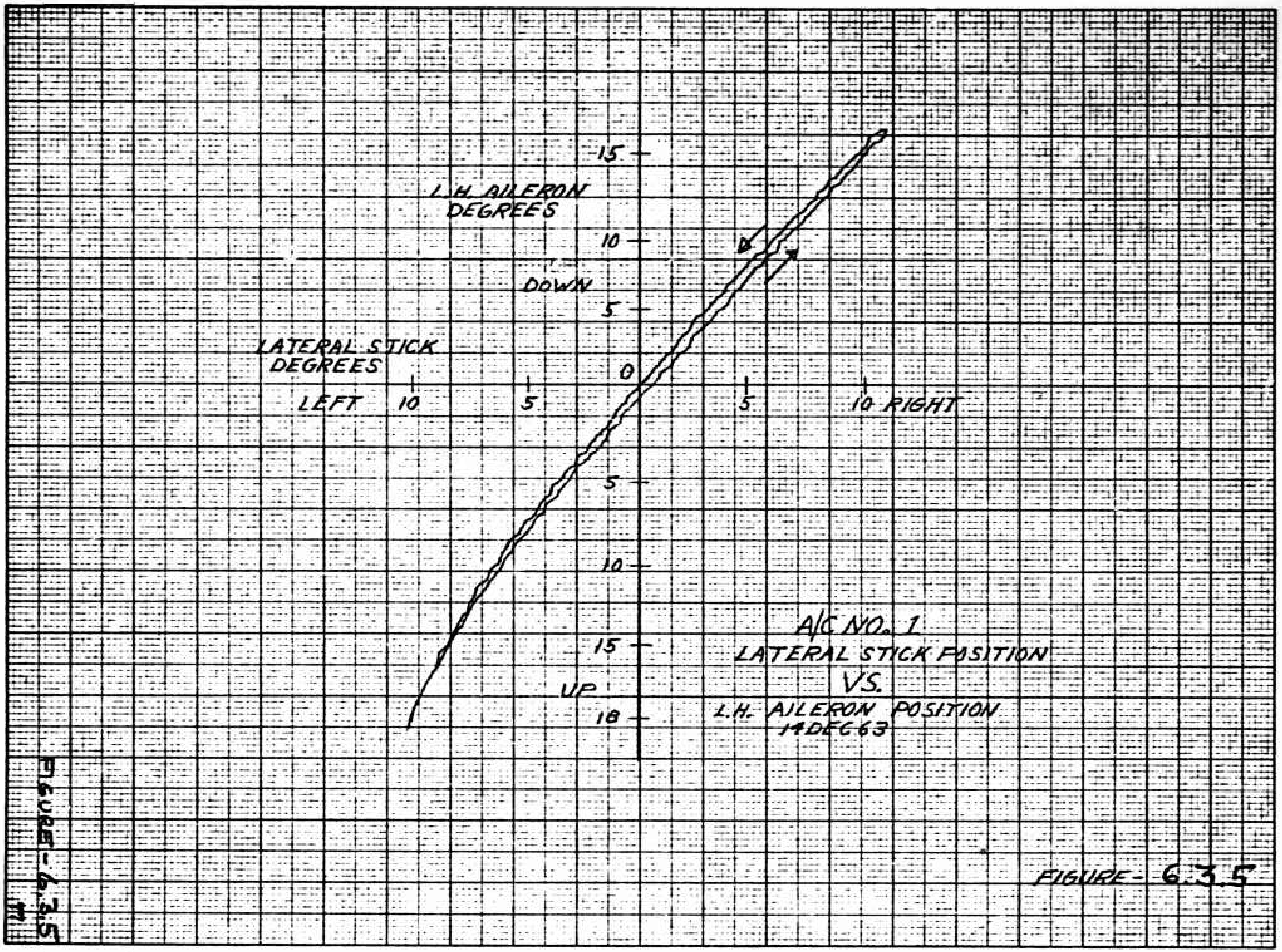
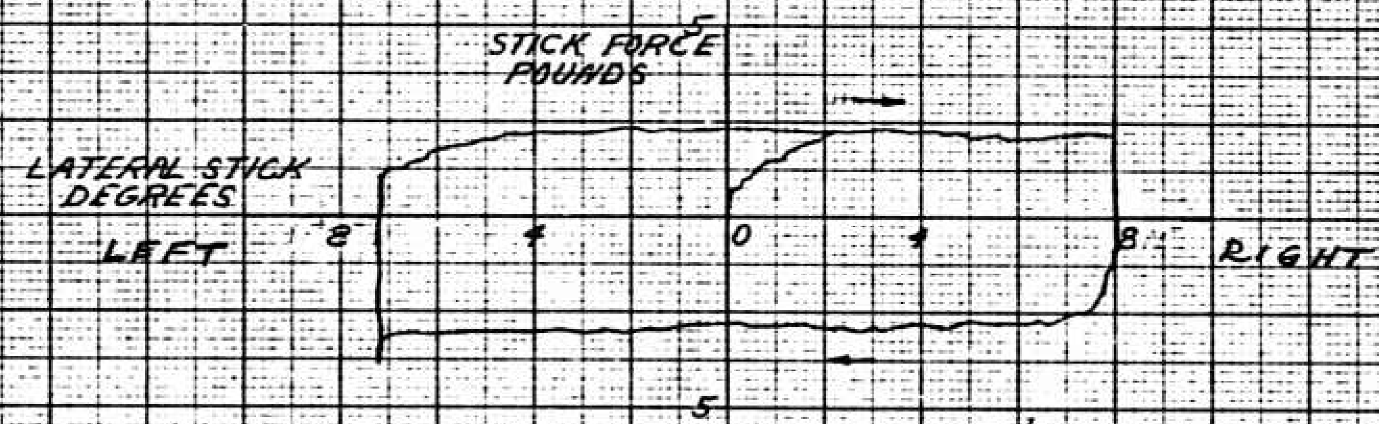

L.H. AILERON DEGREES
15
10
DOWN
5
0
5
10
15
UP
18
LATERAL STICK DEGREES
LEFT
10
5
5
10 RIGHT
A/C NO. 1
LATERAL STICK POSITION
VS.
L.H. AILERON POSITION
14 DEC 63

FIGURE- 6.3.5



A/C NO. 1
LATERAL STICK POSITION
VS.
LATERAL STICK FORCE
CTOL MODE
14 DEC. 63

FIGURE- 6.3.6

FIGURE-6.3.6

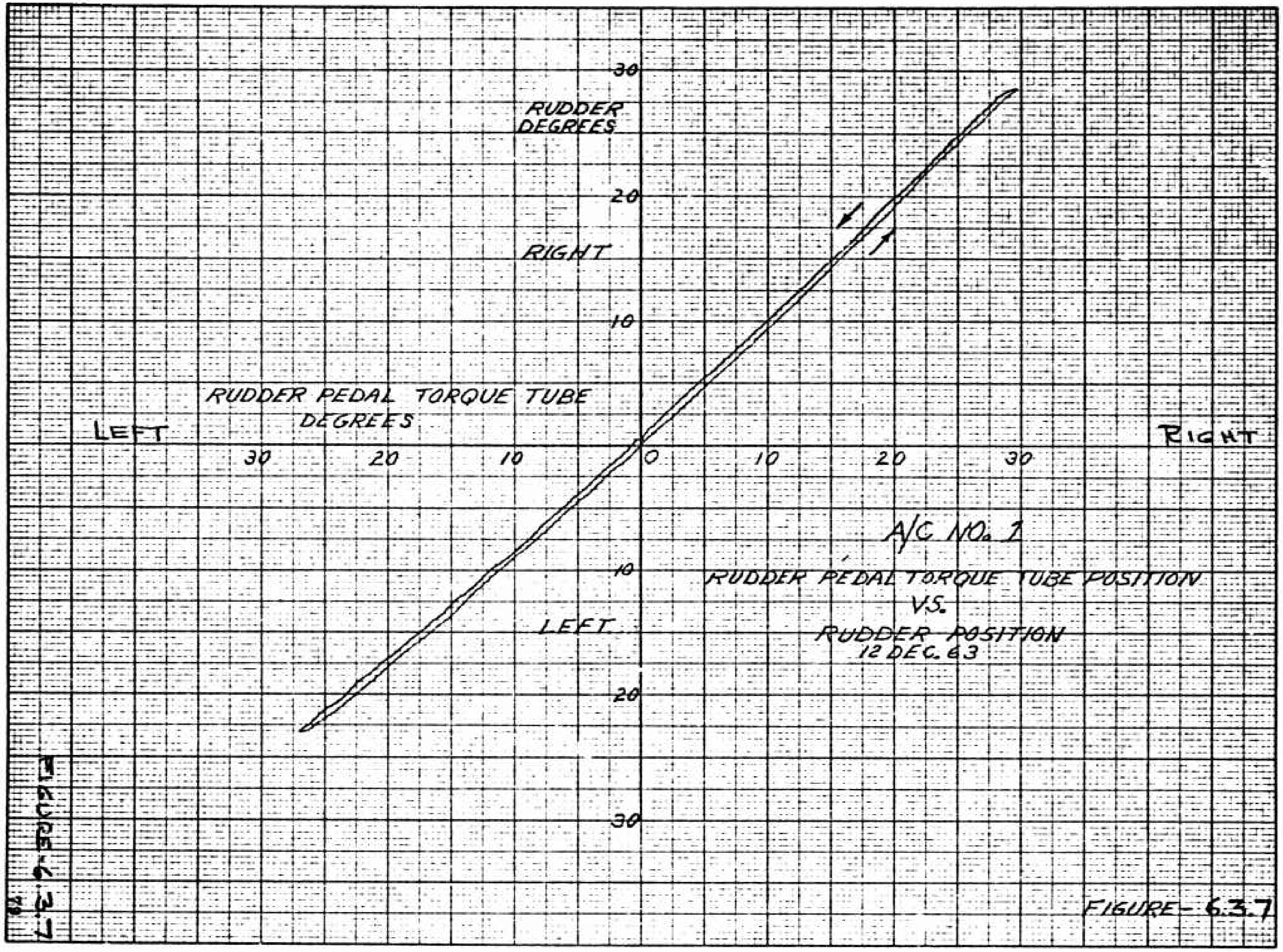
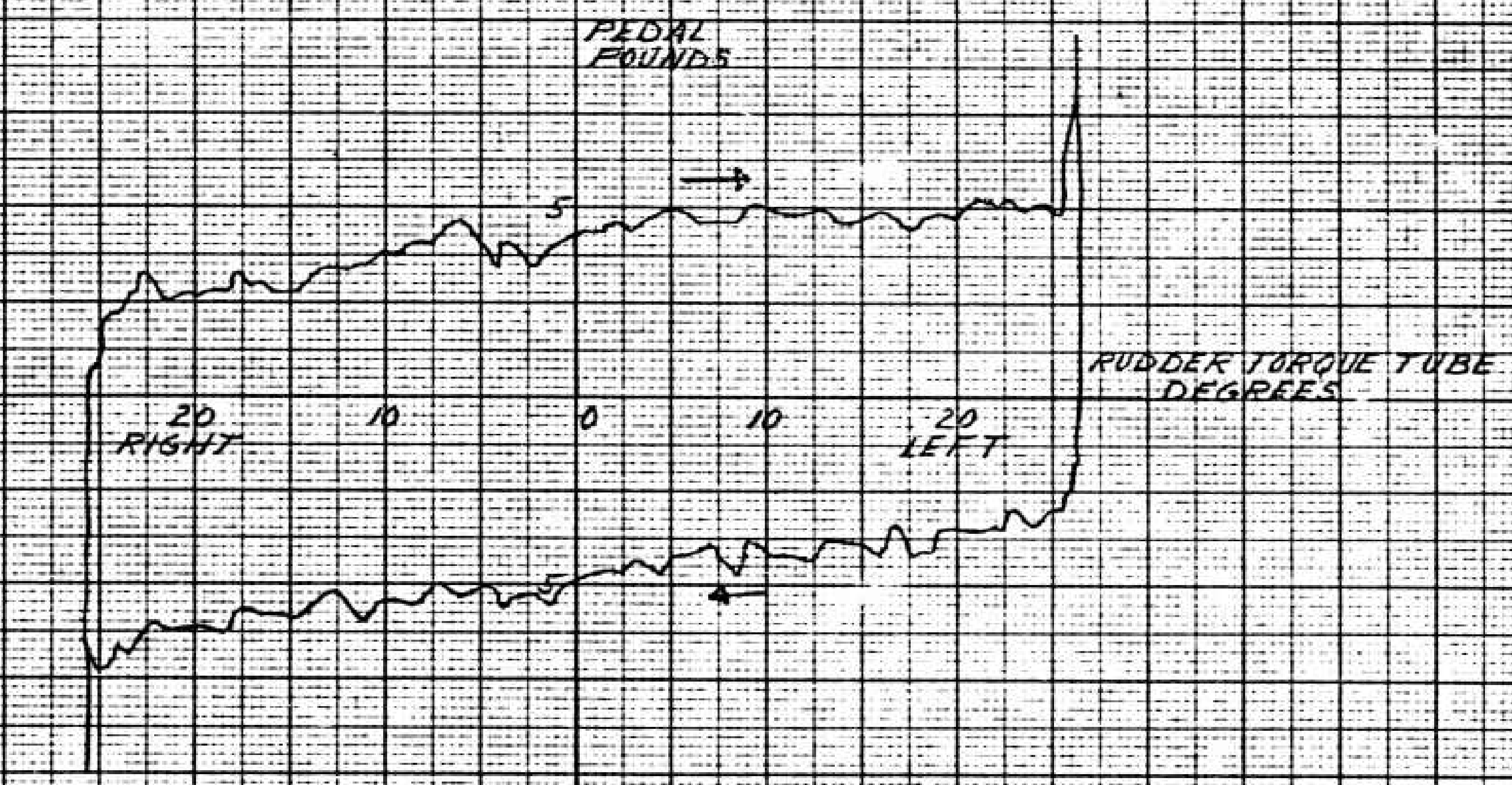


FIGURE-6.3.7

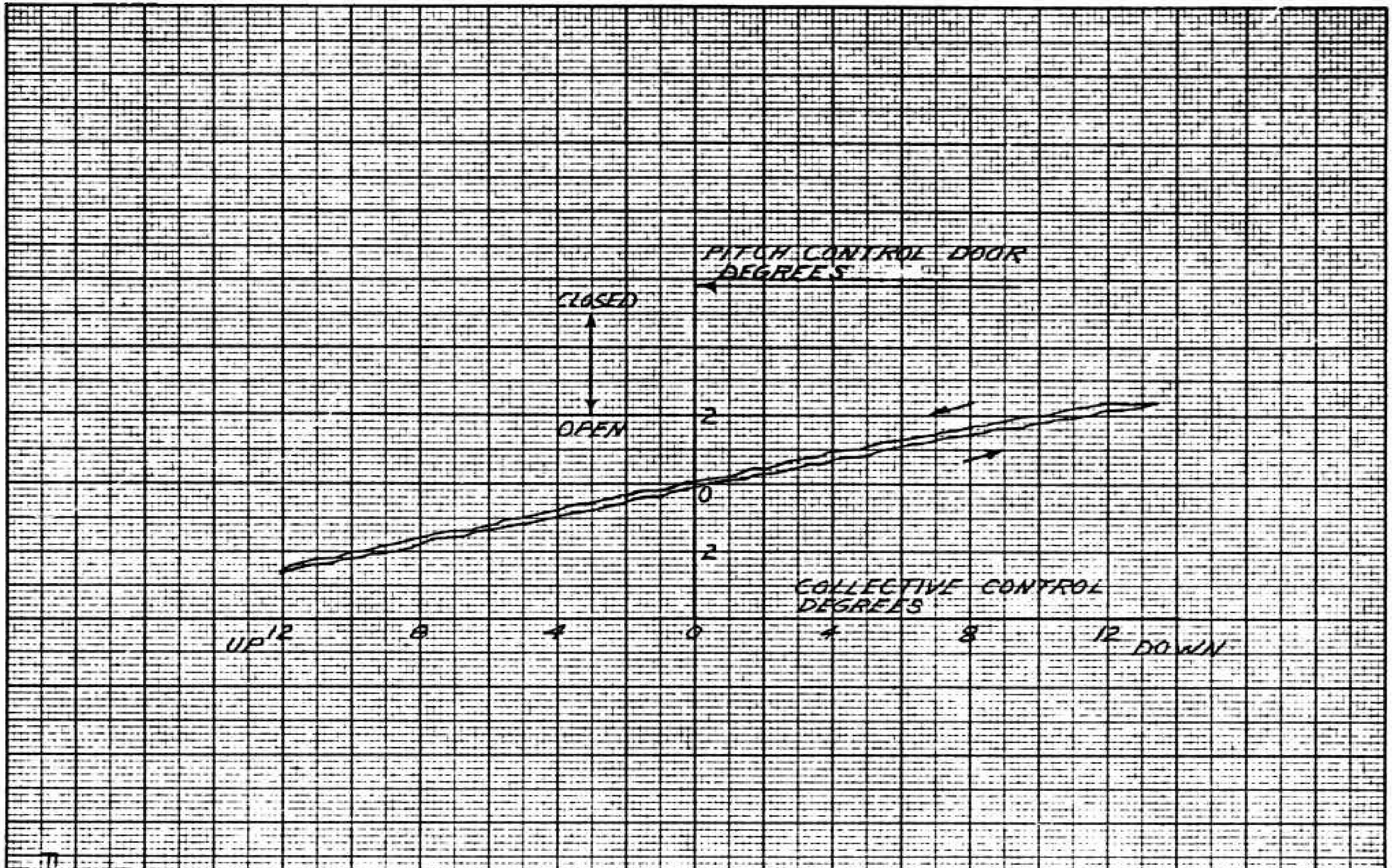
FIGURE-6.3.7



NO. 1 A/C
RUDDER PEDAL POSITION
VS.
PEDAL FORCE, CTOL MODE
10 FEB. 64

FIGURE-6.38

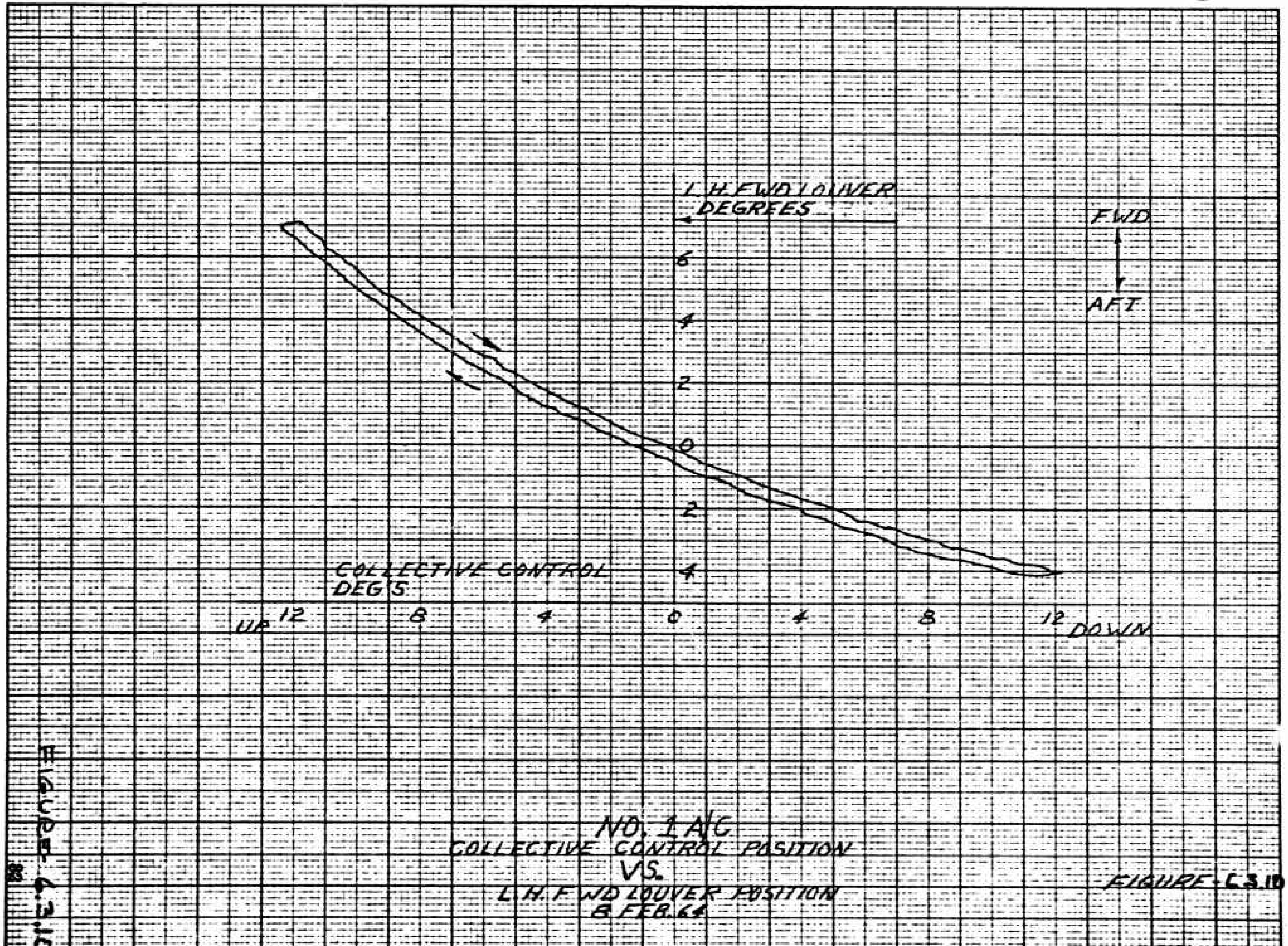
FIGURE-6.38



NO. 1 A/C
COLLECTIVE CONTROL POSITION
VS.
PITCH CONTROL DOOR POSITION
8 FEB 64

FIGURE- 6.3.9

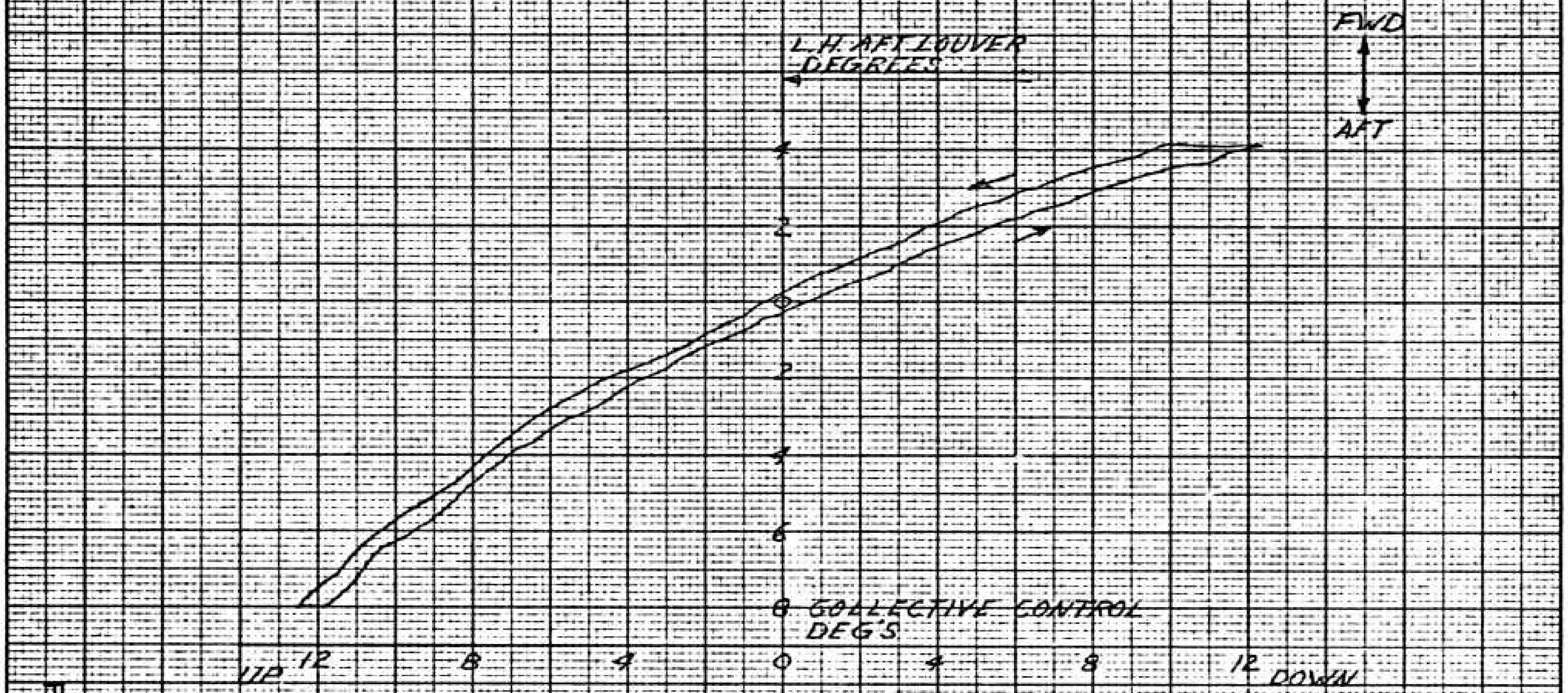
FIGURE-6.3.9



NO. 1 A/C
COLLECTIVE CONTROL POSITION
VS.
L.H. FWD LOUVER POSITION
8 FEB. 64

FIGURE 6.3.10

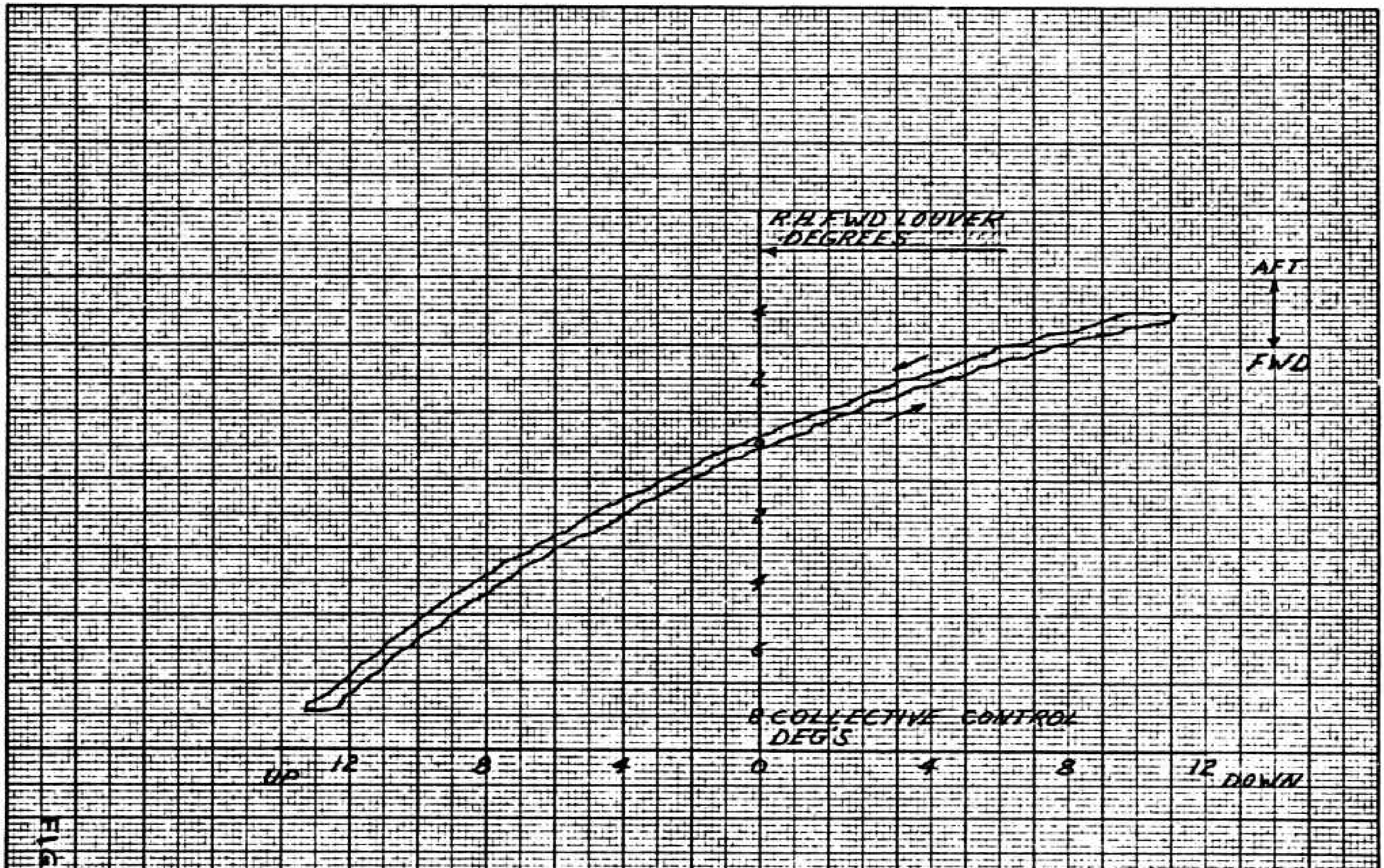
FIGURE 6.3.10



NO. 1 A/C
COLLECTIVE CONTROL POSITION
VS.
L.H. AFT LOUVER POSITION
8 FEB. 64

FIGURE 6.3.11

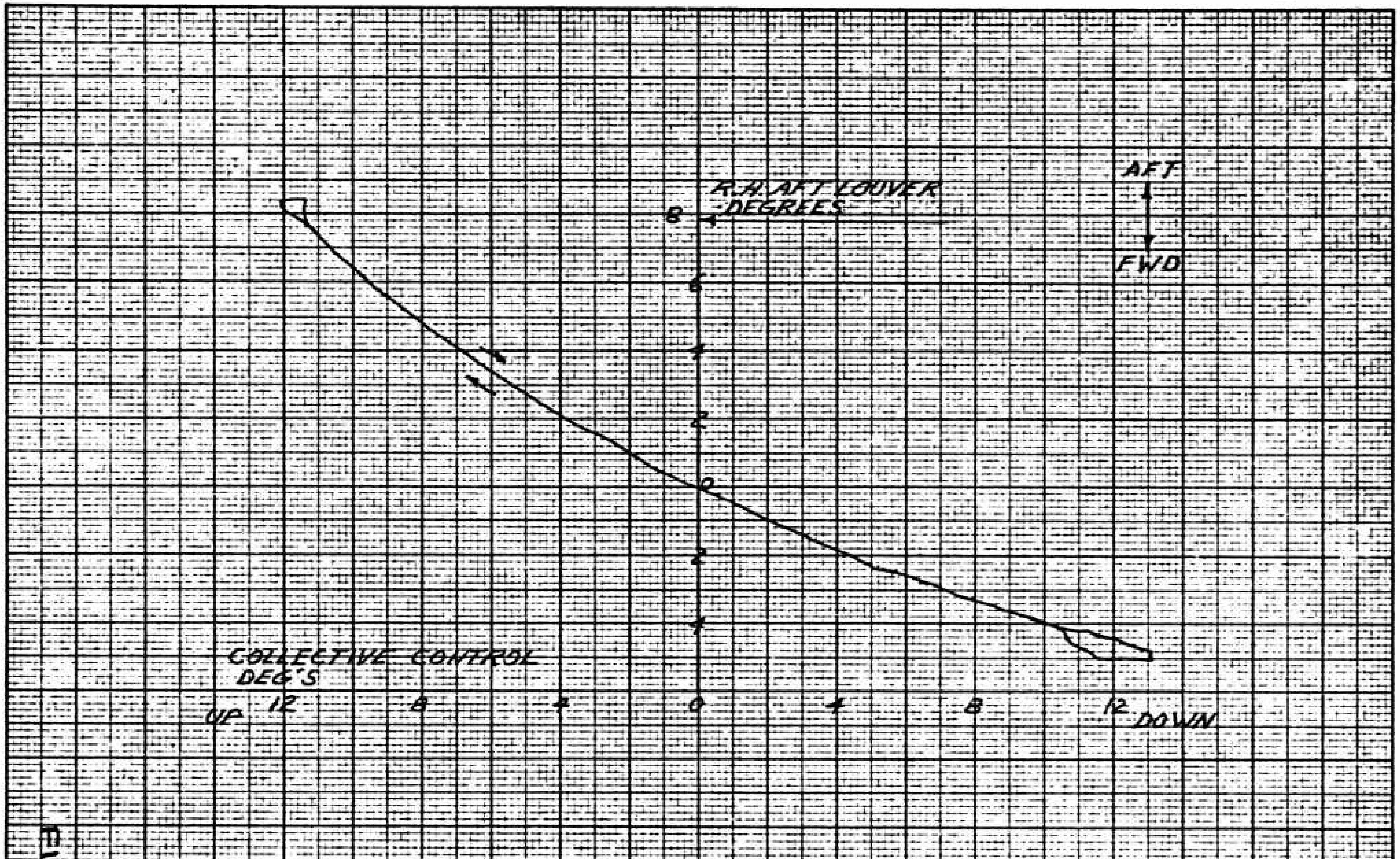
FIGURE-6.3.11



NO. 1 A/C
COLLECTIVE CONTROL POSITION
VS.
R.H. FWD LOUVER POSITION
8 FEB. 64

FIGURE-6.3.12

FIGURE-6.3.12



NO. 1 A/C
COLLECTIVE CONTROL POSITION
VS.
R.H. AFT LOUVER POSITION
8 FEB. 64

FIGURE-6.3.13

FIGURE-6.3.13

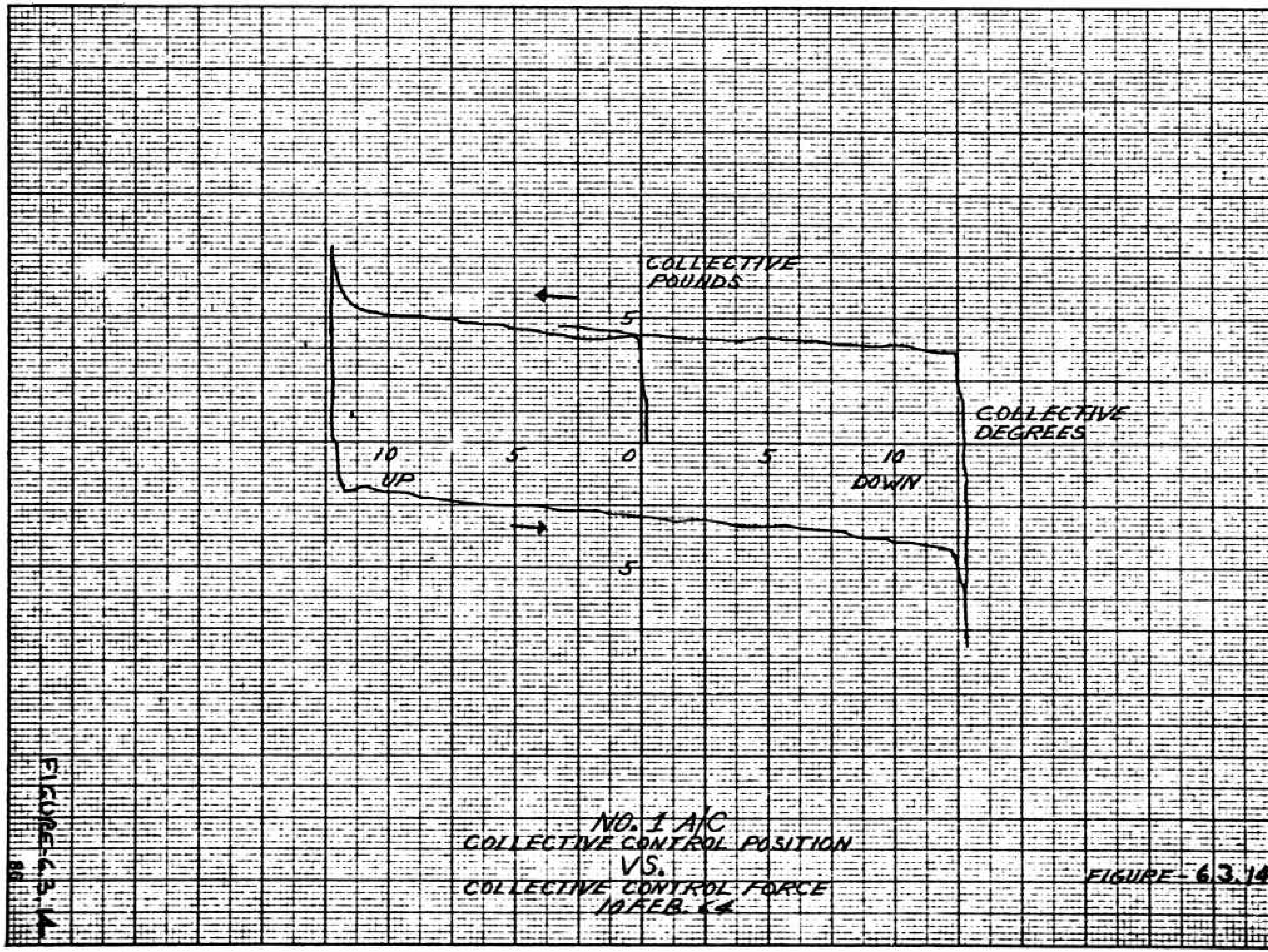

COLLECTIVE
POUNDS
5
COLLECTIVE
DEGREES
10
5
0
5
10
UP
DOWN
5
NO. 1 A/C
COLLECTIVE CONTROL POSITION
VS.
COLLECTIVE CONTROL FORCE
10 FEB. 64
FIGURE 6.3.14

FIGURE-6.3.14

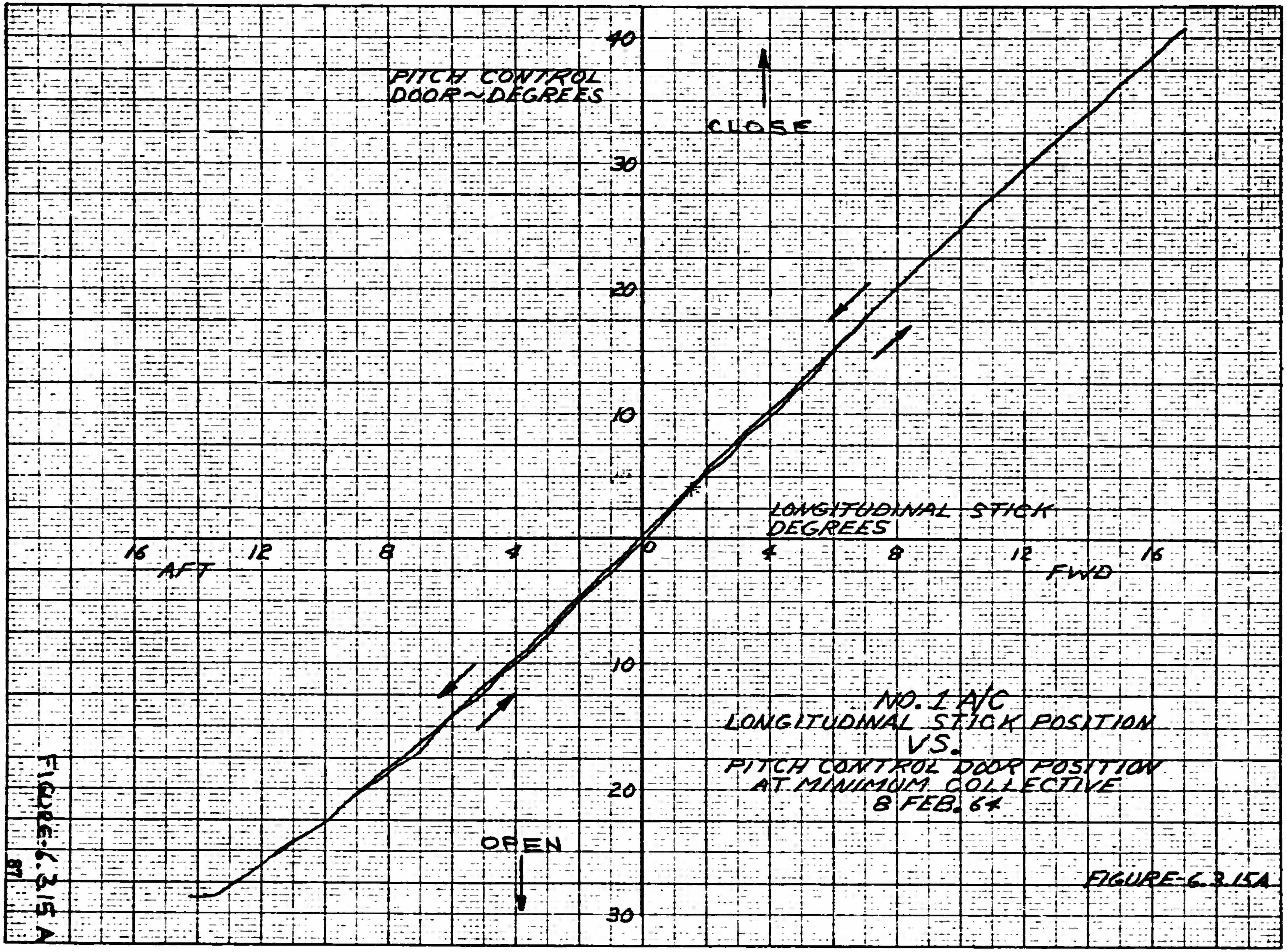


FIGURE-6.3.15 A

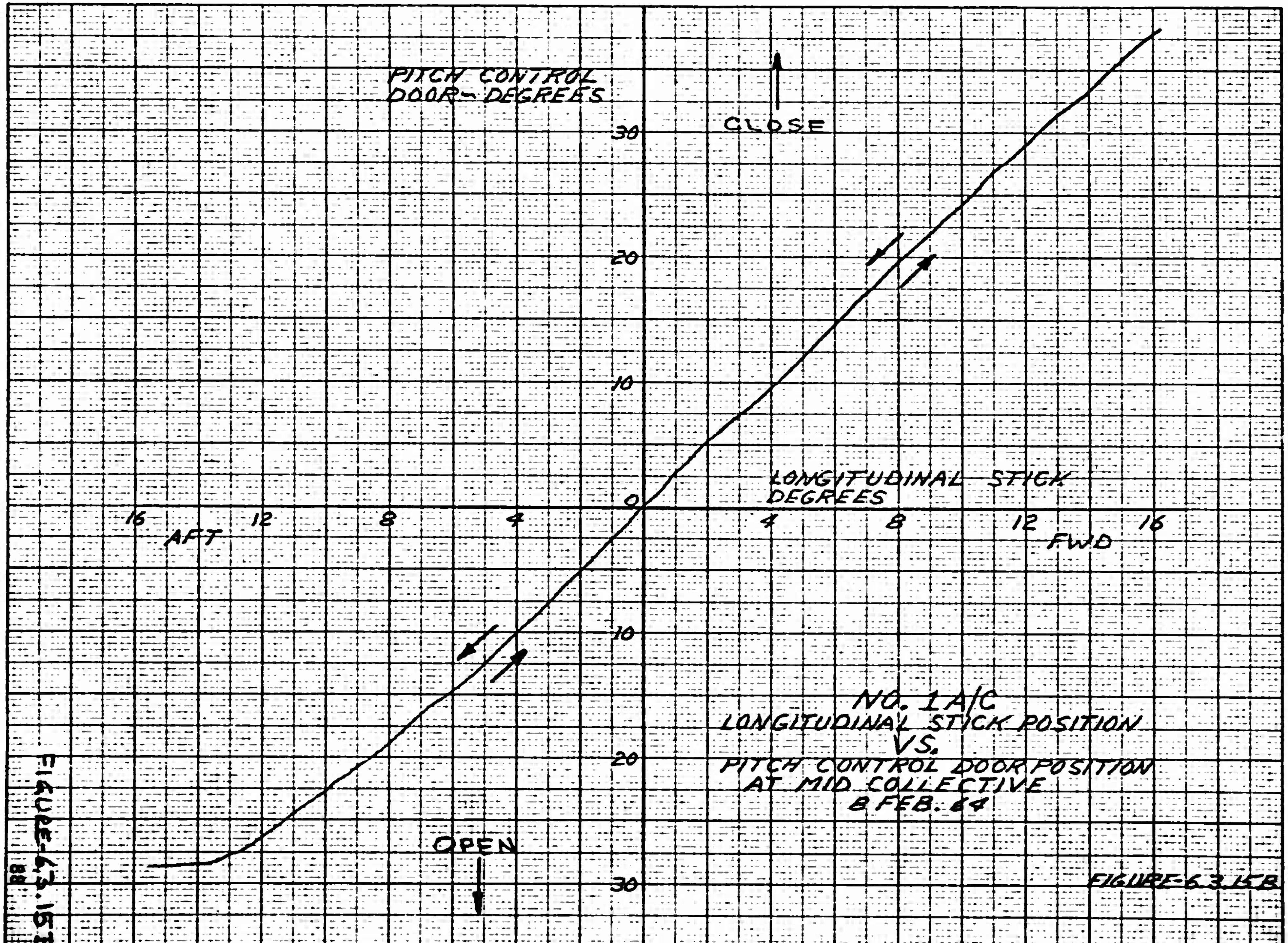


FIGURE-6.3.15B

FIGURE-6.3.15B

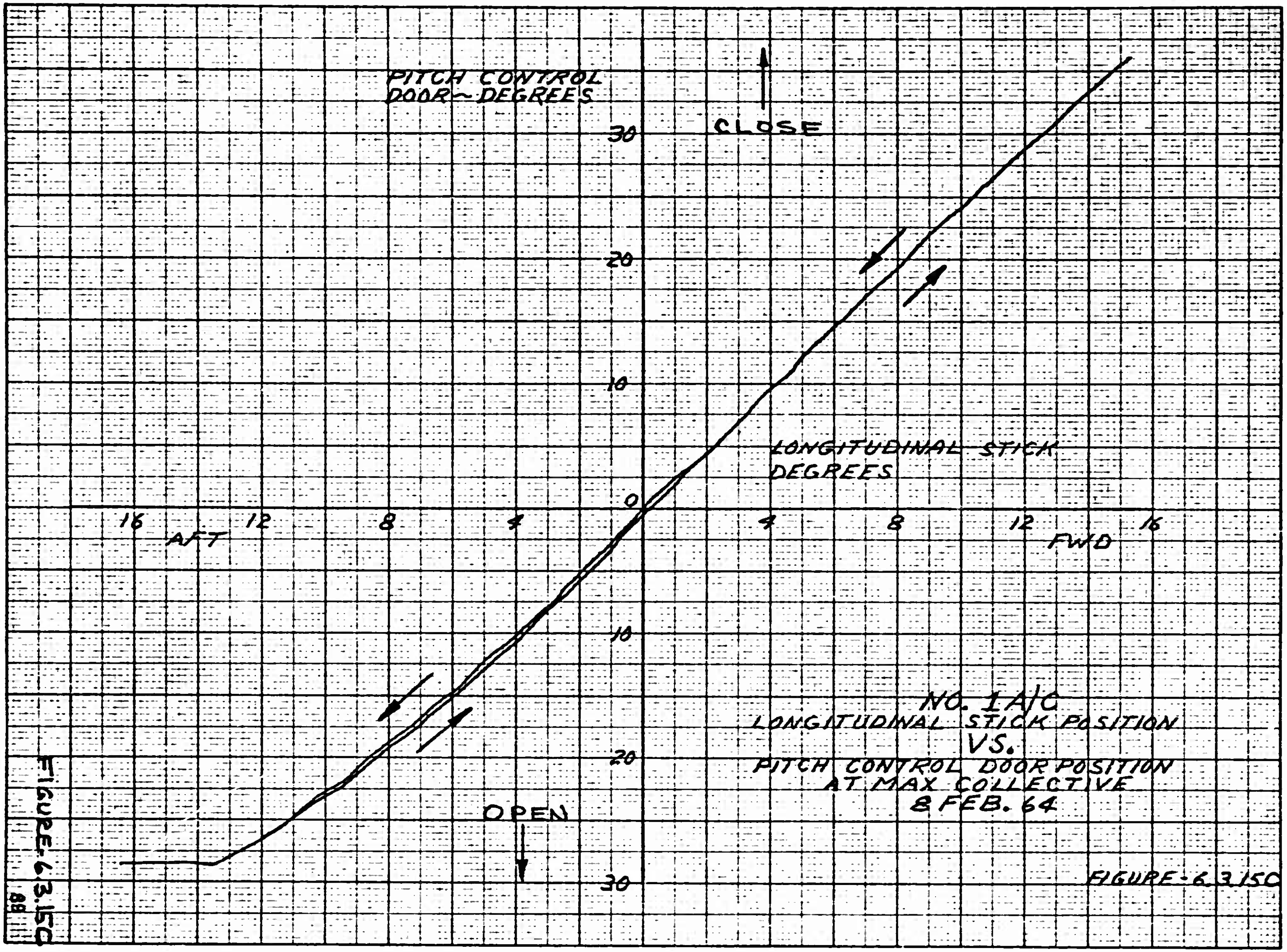
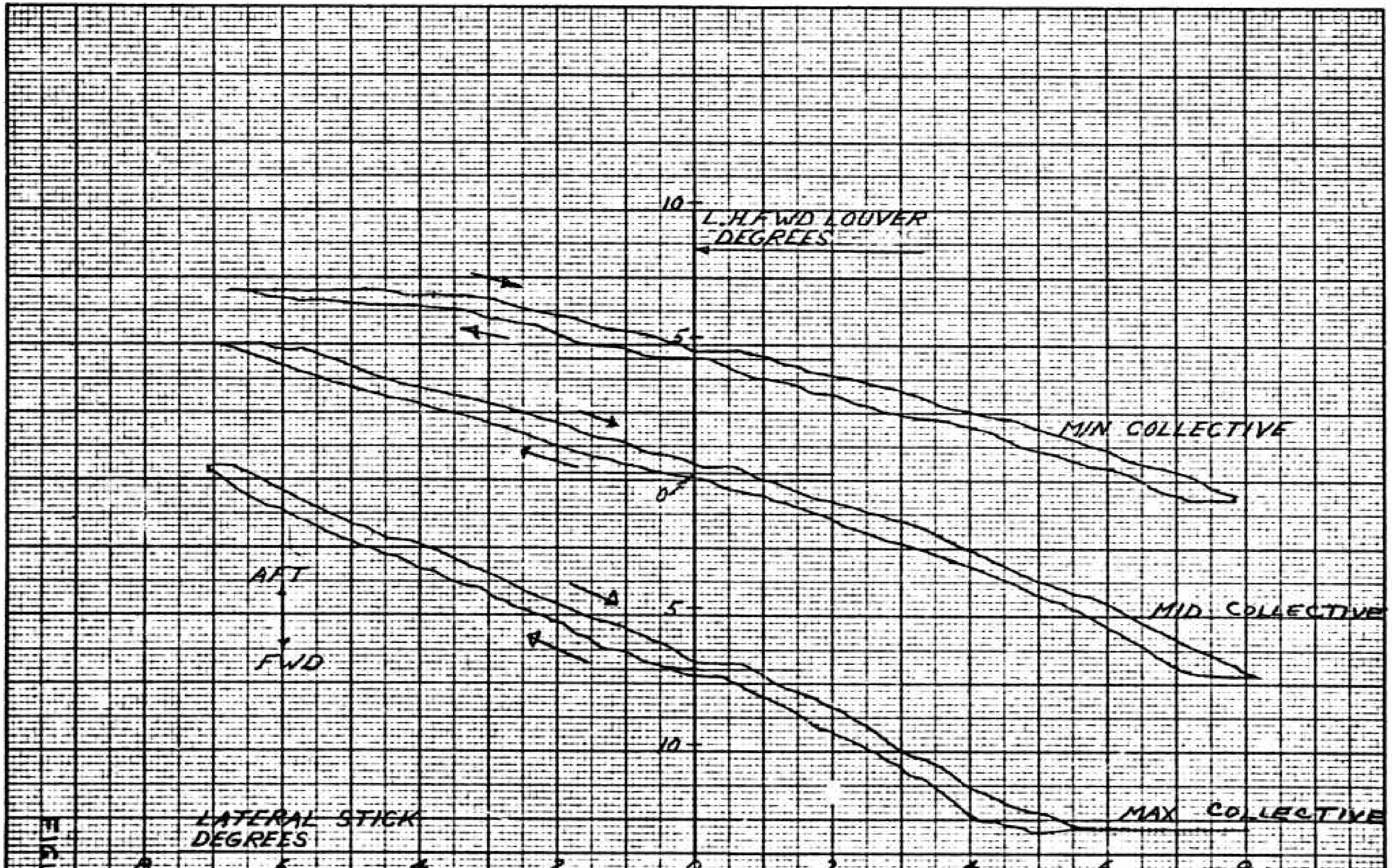


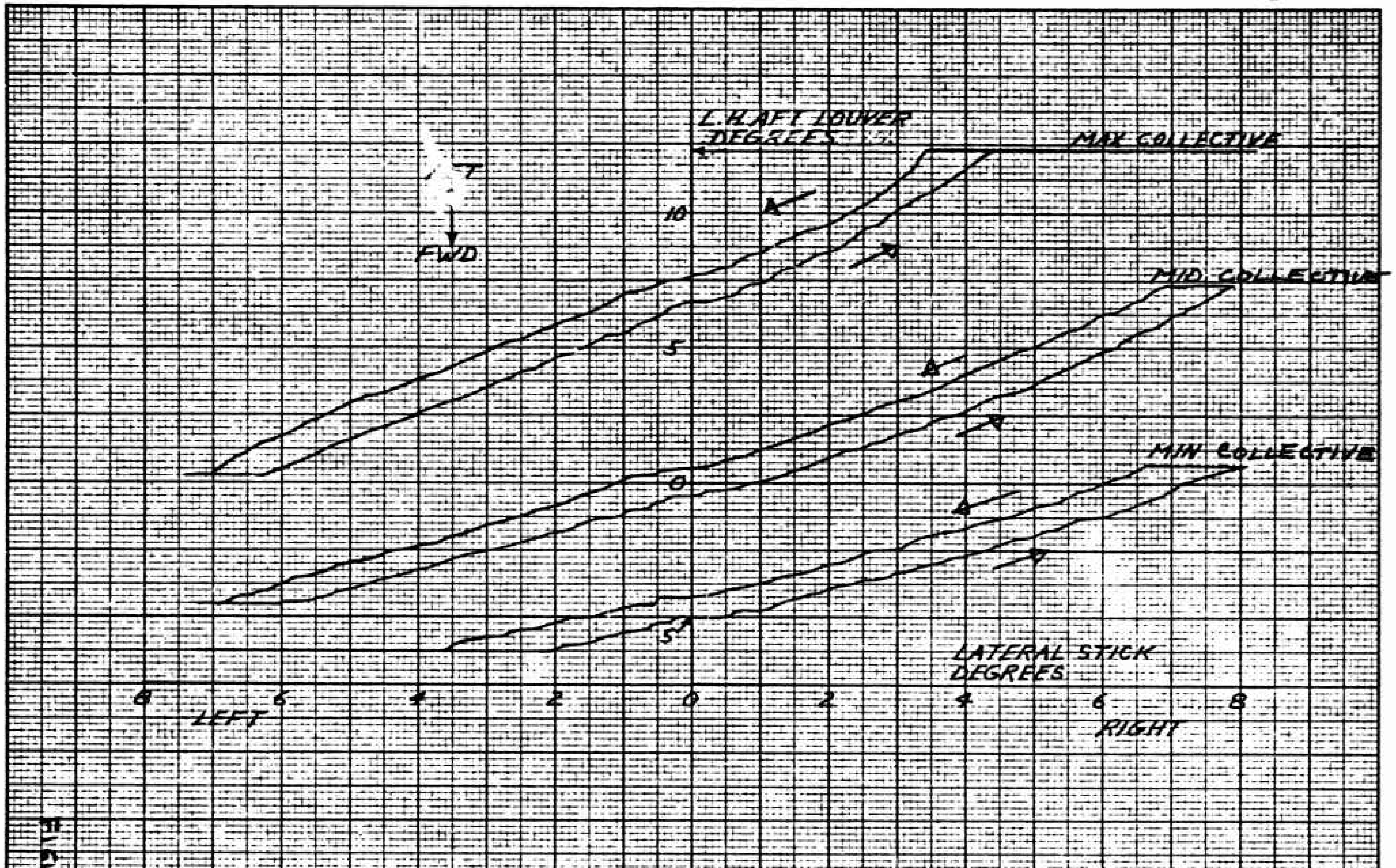
FIGURE 6.3.15C



NO. 1 A/C
LATERAL STICK POSITION
V.S.
L.H. FWD LOUVER POSITION
11 FEB. 64

FIGURE- 6.3.16

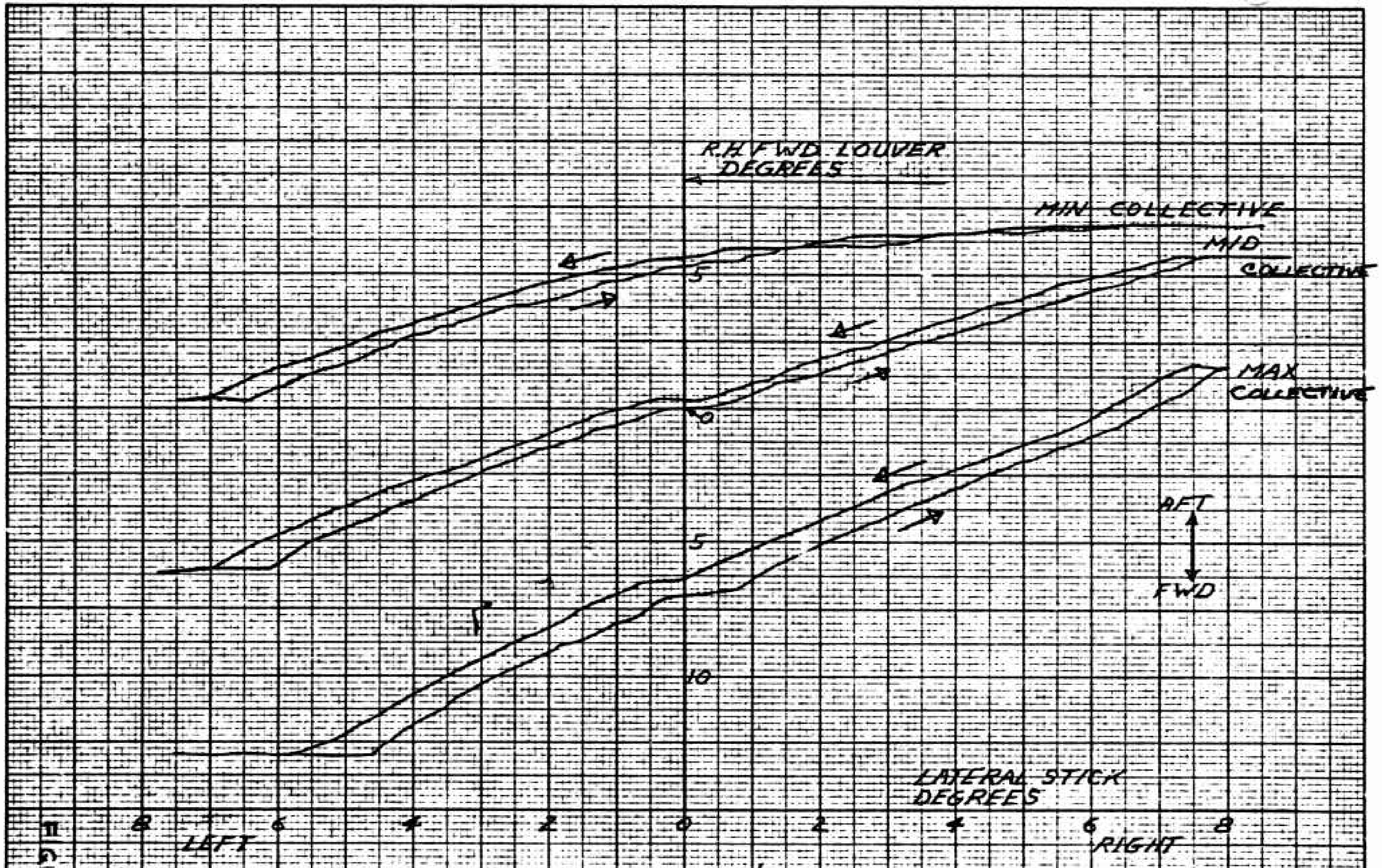
FIGURE-6.3.6



NO. 1 A/C
LATERAL STICK POSITION
VS.
L.H. AFT LOUVER POSITION
11 FEB. 64

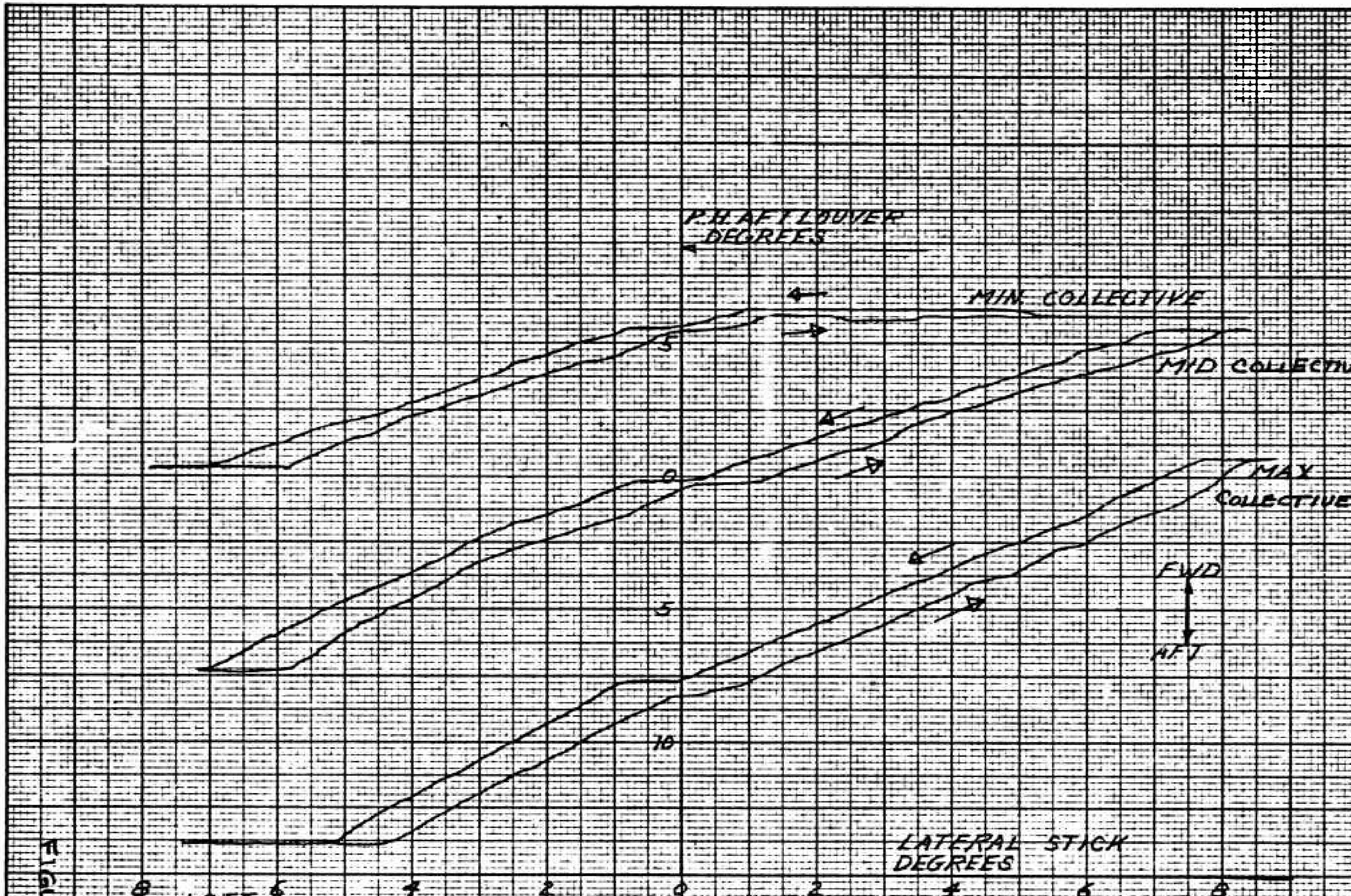
FIGURE-6.3.17

FIGURE-6.3.17



NO. 1 A/C
LATERAL STICK POSITION
VS.
R.H. FWD LOUVER POSITION
12 FEB 64

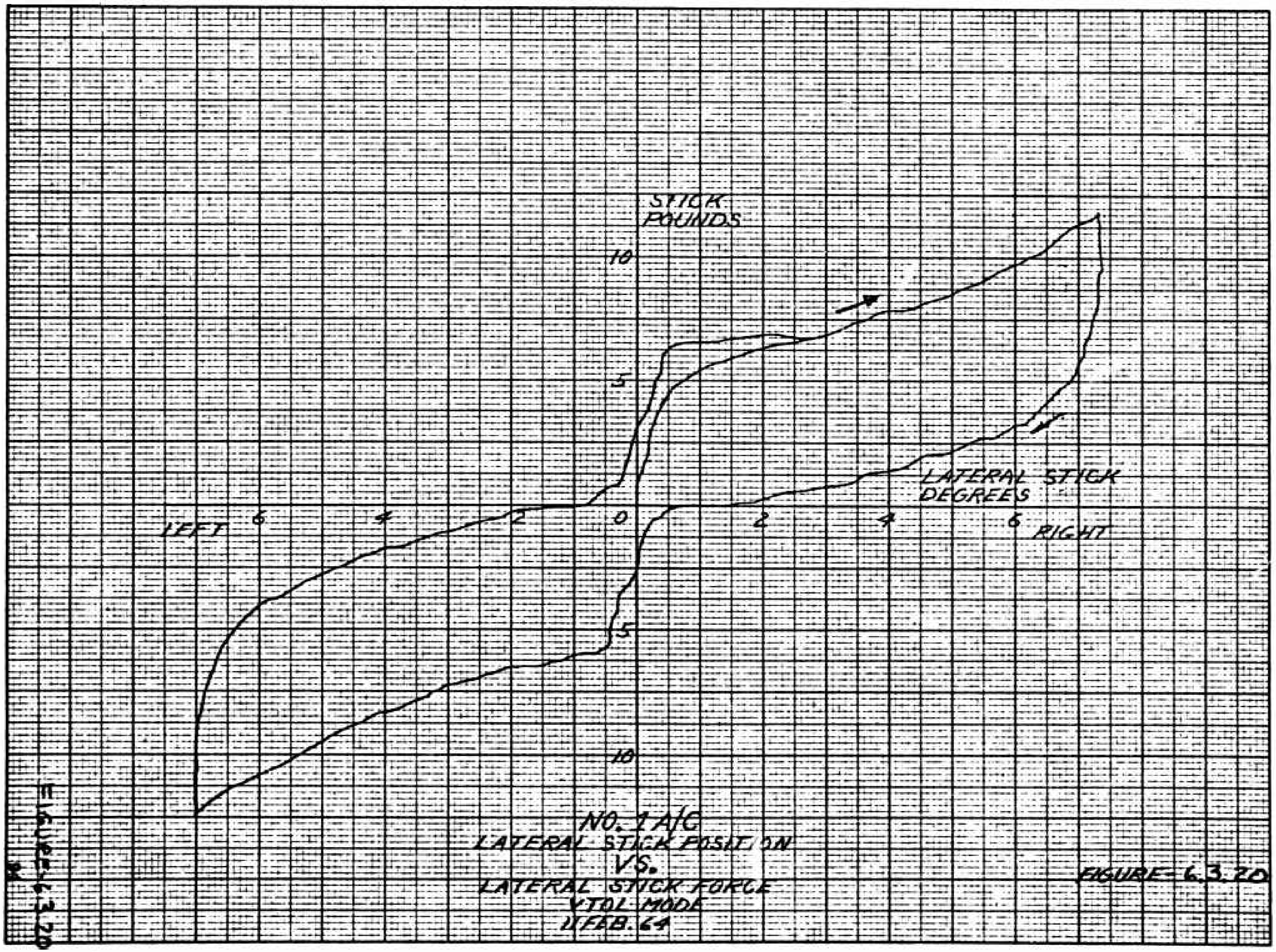
FIGURE-6.3.18



NO. 1 A/C
LATERAL STICK POSITION
VS.
R.H. AFT LOUVER POSITION
12 FEB 64

FIGURE 6.3.19

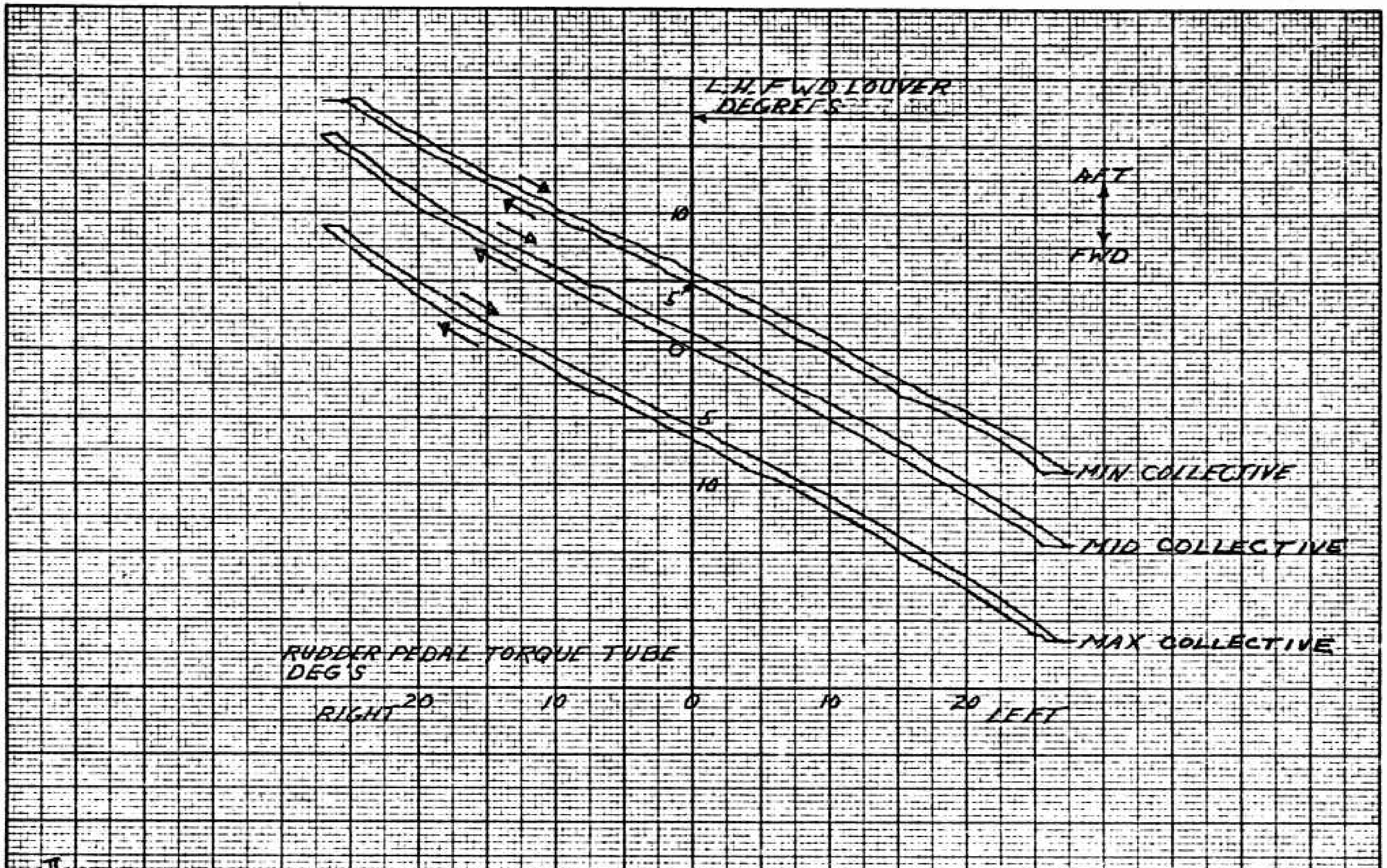
FIGURE 6.3.19



NO. 1 A/C
LATERAL STICK POSITION
VS.
LATERAL STICK FORCE
VTOL MODE
11 FEB. 64

FIGURE-6.3.20

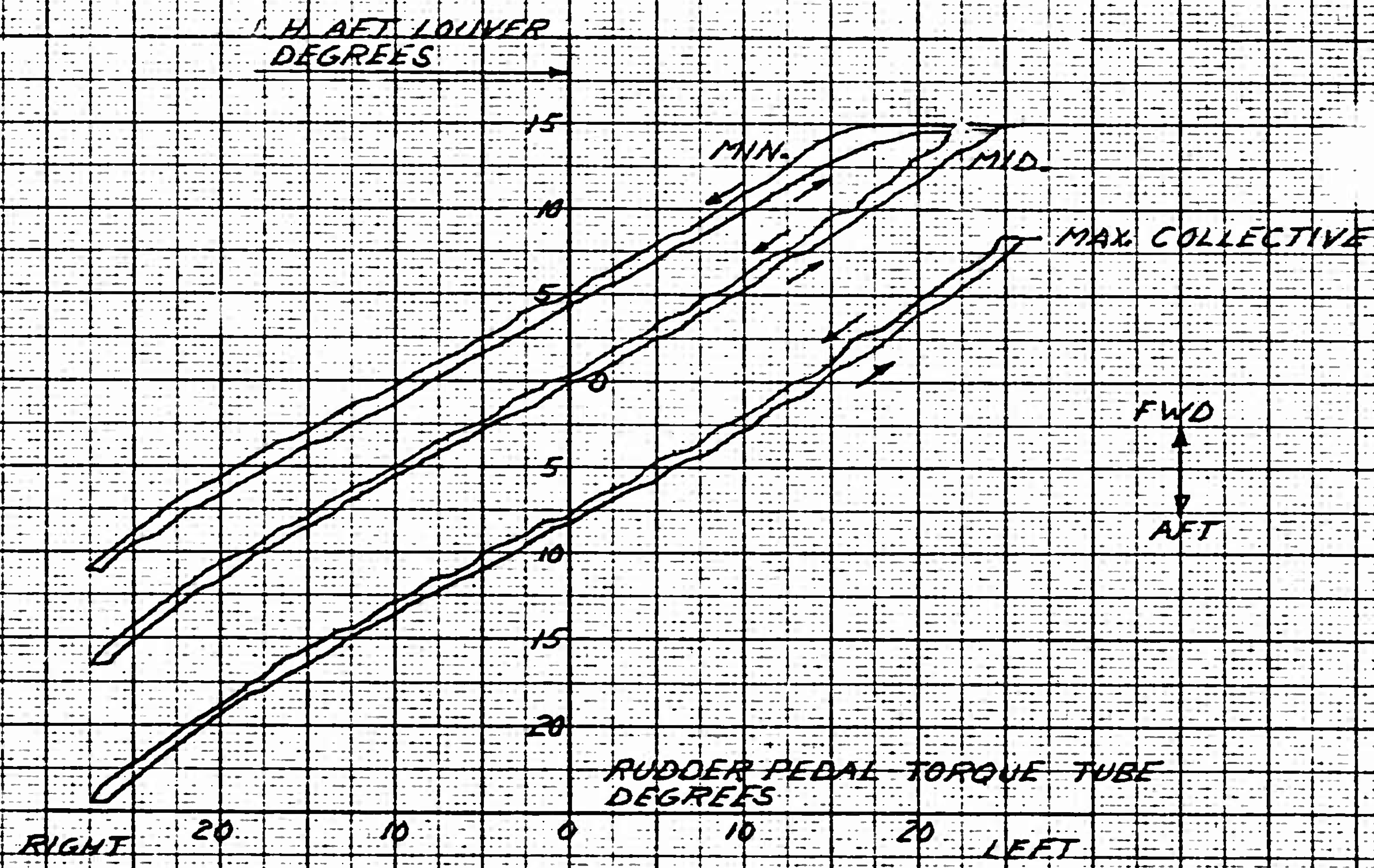
FIGURE-6.3.20



NO. 1 A/C
RUDDER PEDAL TORQUE TUBE POSITION
VS.
L.H. FWD LOUVER POSITION
10 FEB 64

FIGURE-6.3.21

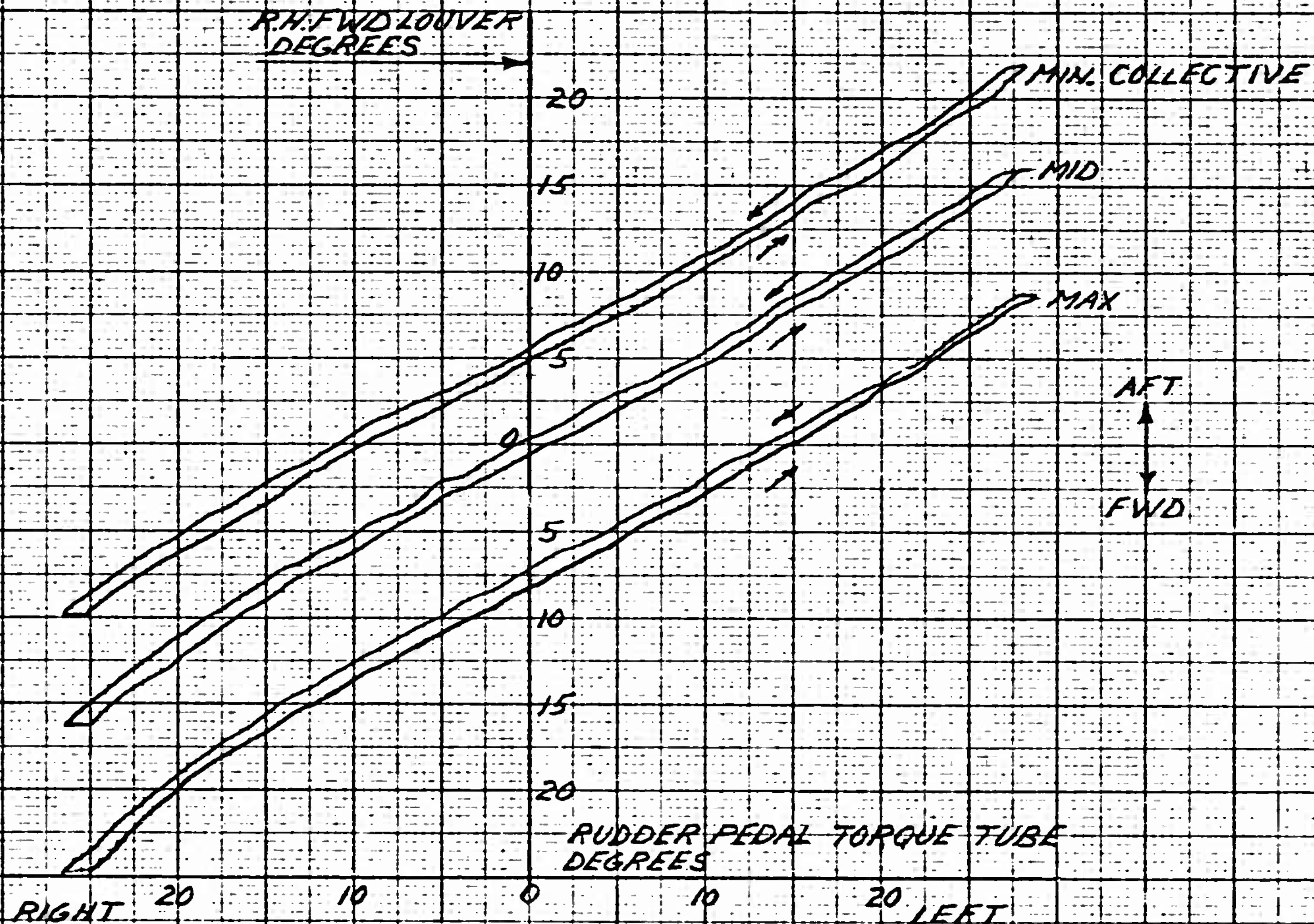
FIGURE-6.3.21



NO. 1 A/C
RUDDER PEDAL TORQUE TUBE POSITION
VS.
L.H. AFT LOUVER POSITION
10 FEB. 64

FIGURE-6.3.22

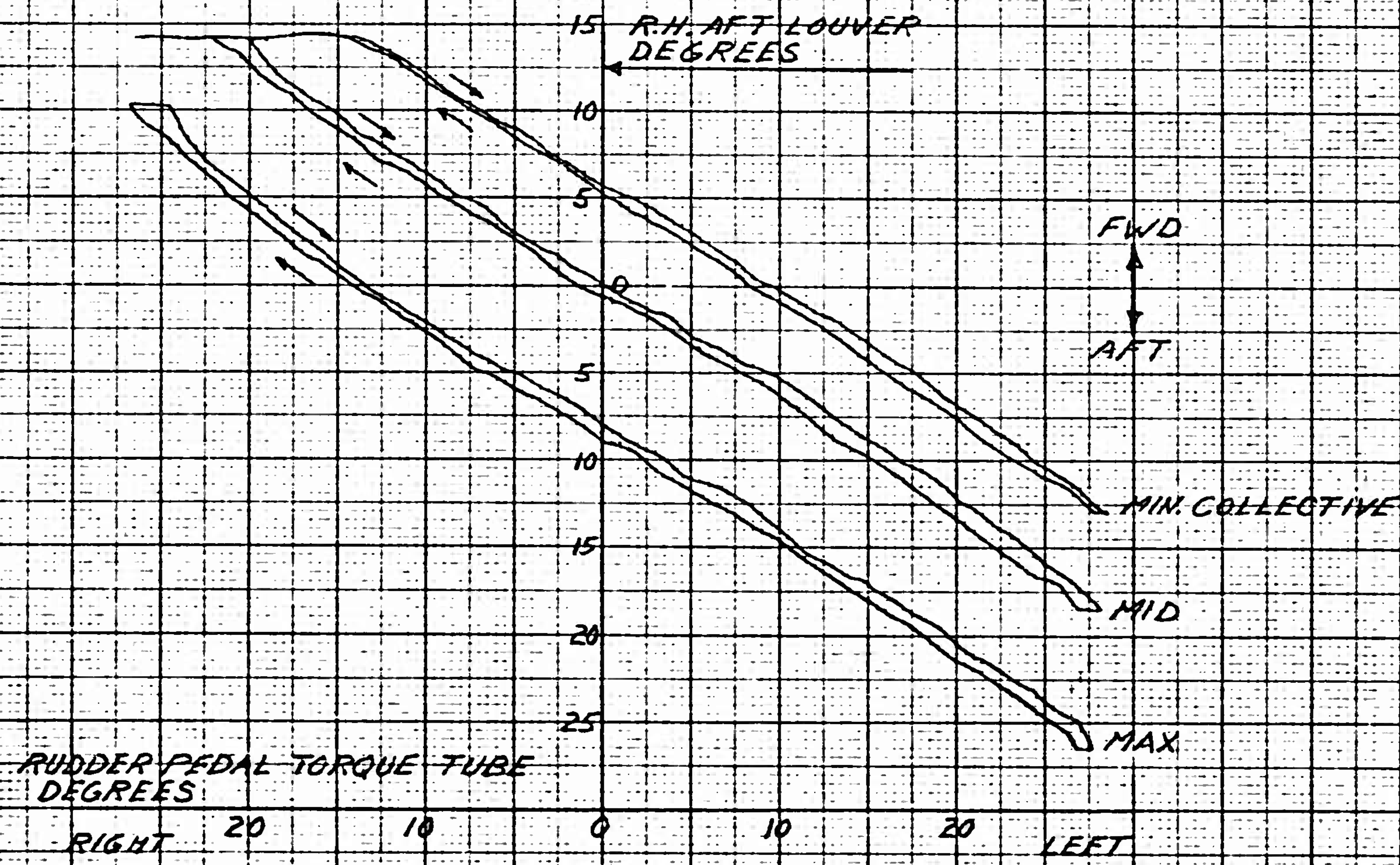
FIGURE 6.3.22



NO. 1 A/C
RUDDER PEDAL TORQUE TUBE POSITION
VS.
R.H. FWD LOUVER POSITION
8 FEB. 64

FIGURE-6.3.23

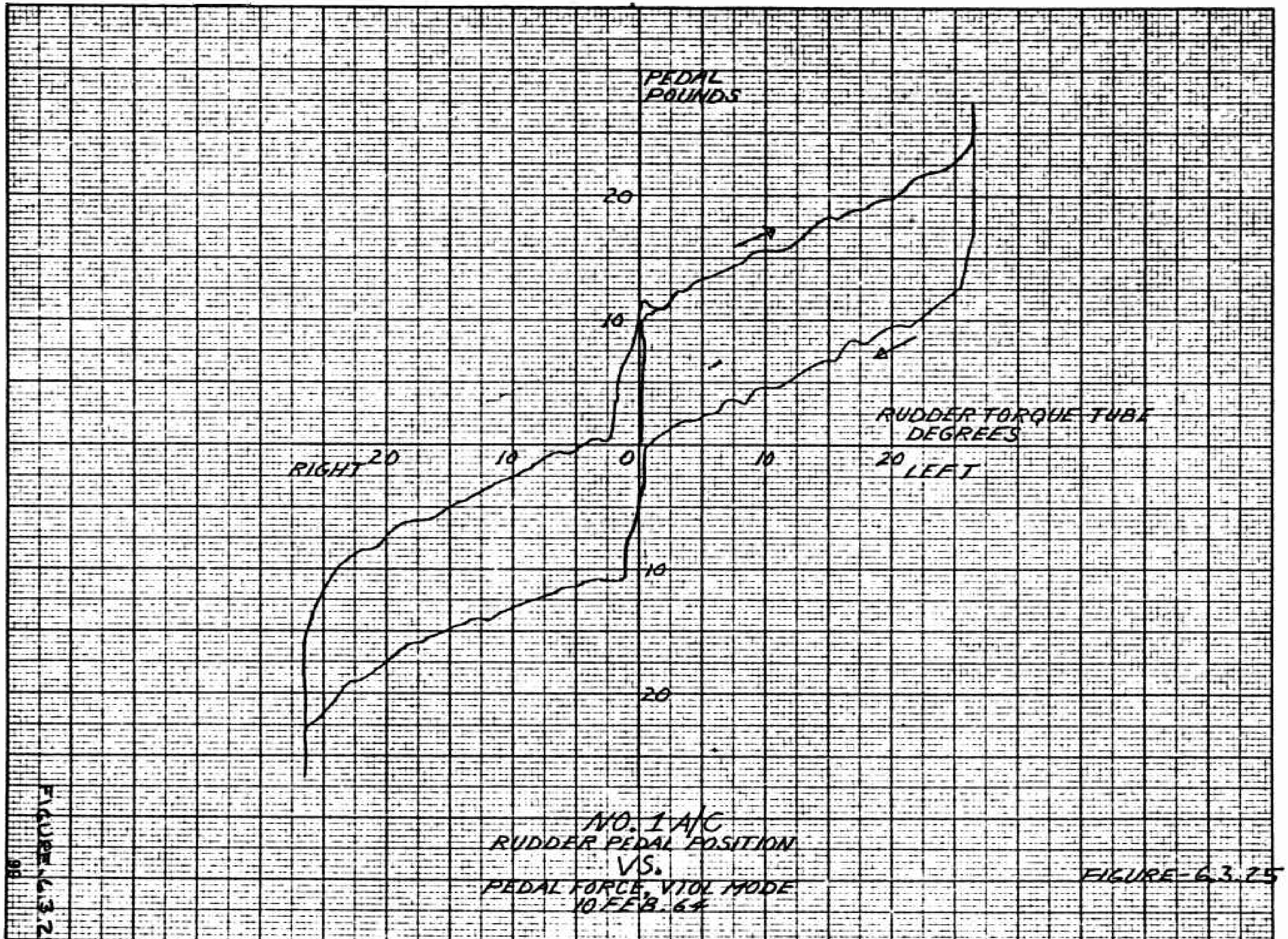
FIGURE 6.3.23



NO. 1 A/C
RUDDER PEDAL TORQUE TUBE POSITION
VS.
R.H. AFT LOUVER POSITION
8 FEB 64

FIGURE-6.3.24

FIGURE-6.3.24



NO. 1 A/C
RUDDER PEDAL POSITION
VS.
PEDAL FORCE, VTOL MODE
10 FEB. 64

FIGURE 6.3.25

FIGURE 6.3.25

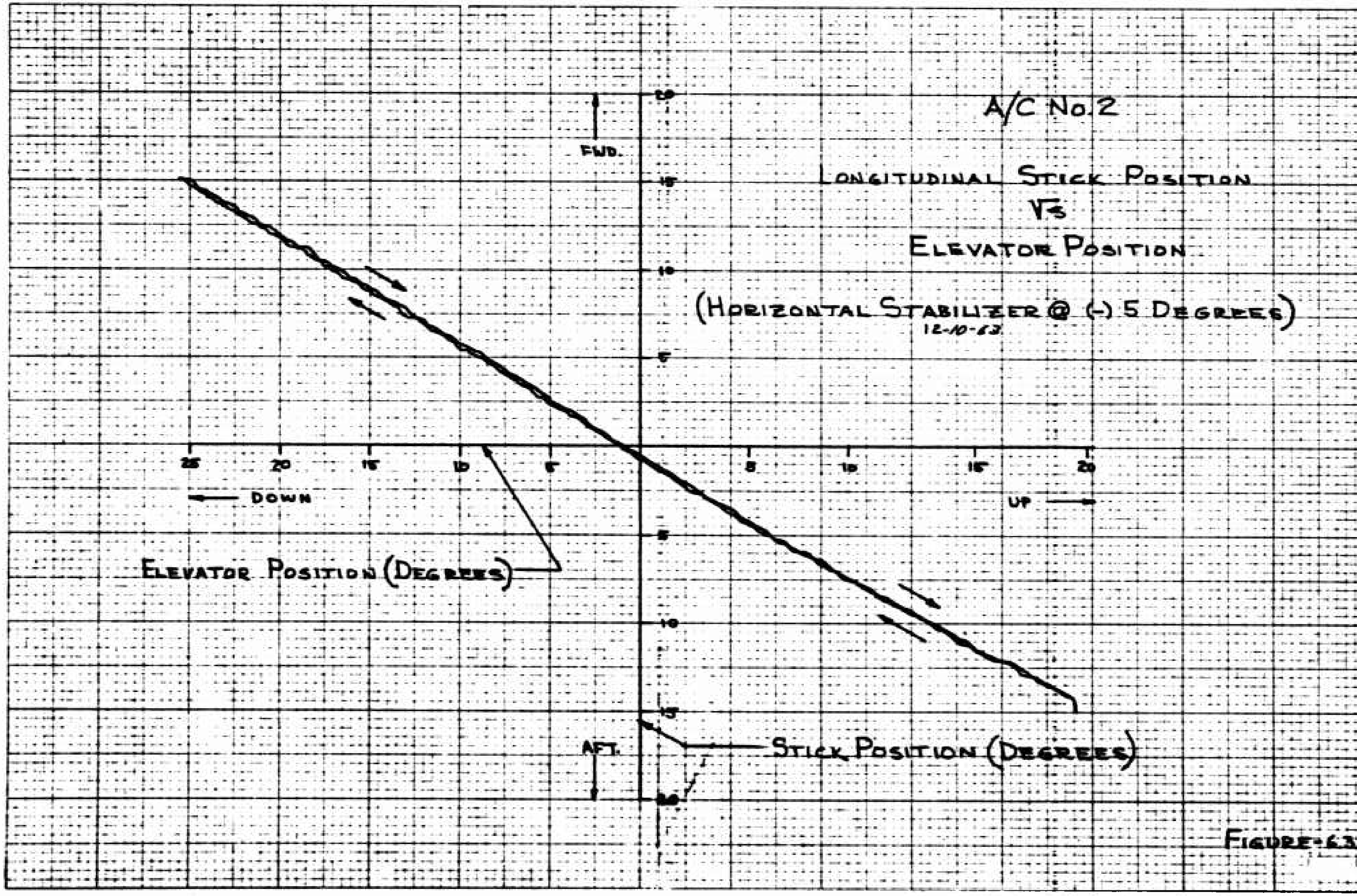

A/C No. 2
Longitudinal Stick Position
Vs
Elevator Position
(Horizontal Stabilizer @ (-) 5 Degrees)
12-10-63
FWD.
AFT.
DOWN
UP
Elevator Position (Degrees)
Stick Position (Degrees)
25
20
15
10
5
5
10
15
20

Figure-6326

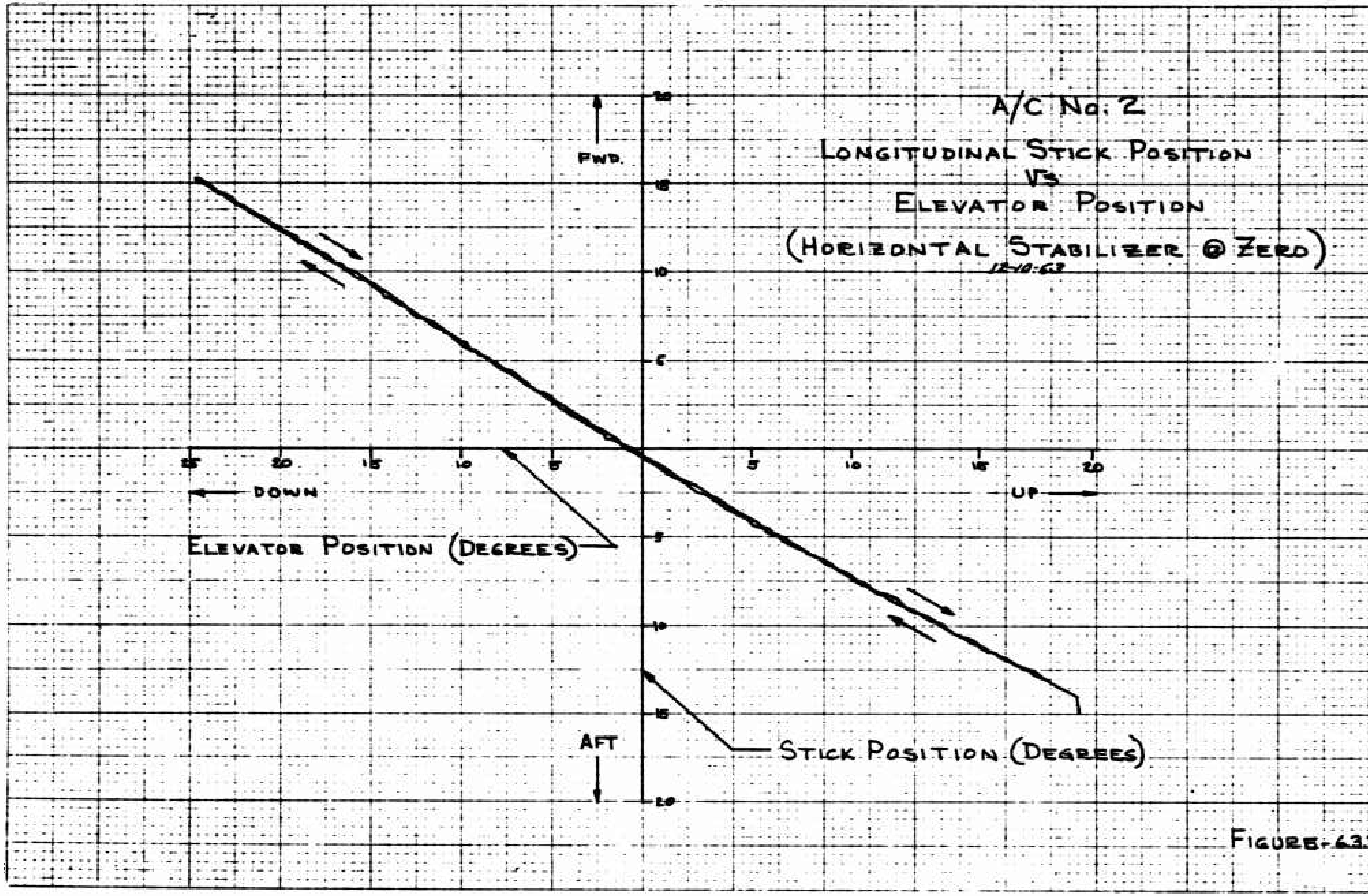

A/C No. 2
LONGITUDINAL STICK POSITION
VS
ELEVATOR POSITION
(HORIZONTAL STABILIZER @ ZERO)
12-10-63
FWD.
AFT
DOWN
UP
ELEVATOR POSITION (DEGREES)
STICK POSITION (DEGREES)
25
20
15
10
5
5
10
15
20

FIGURE-63.27

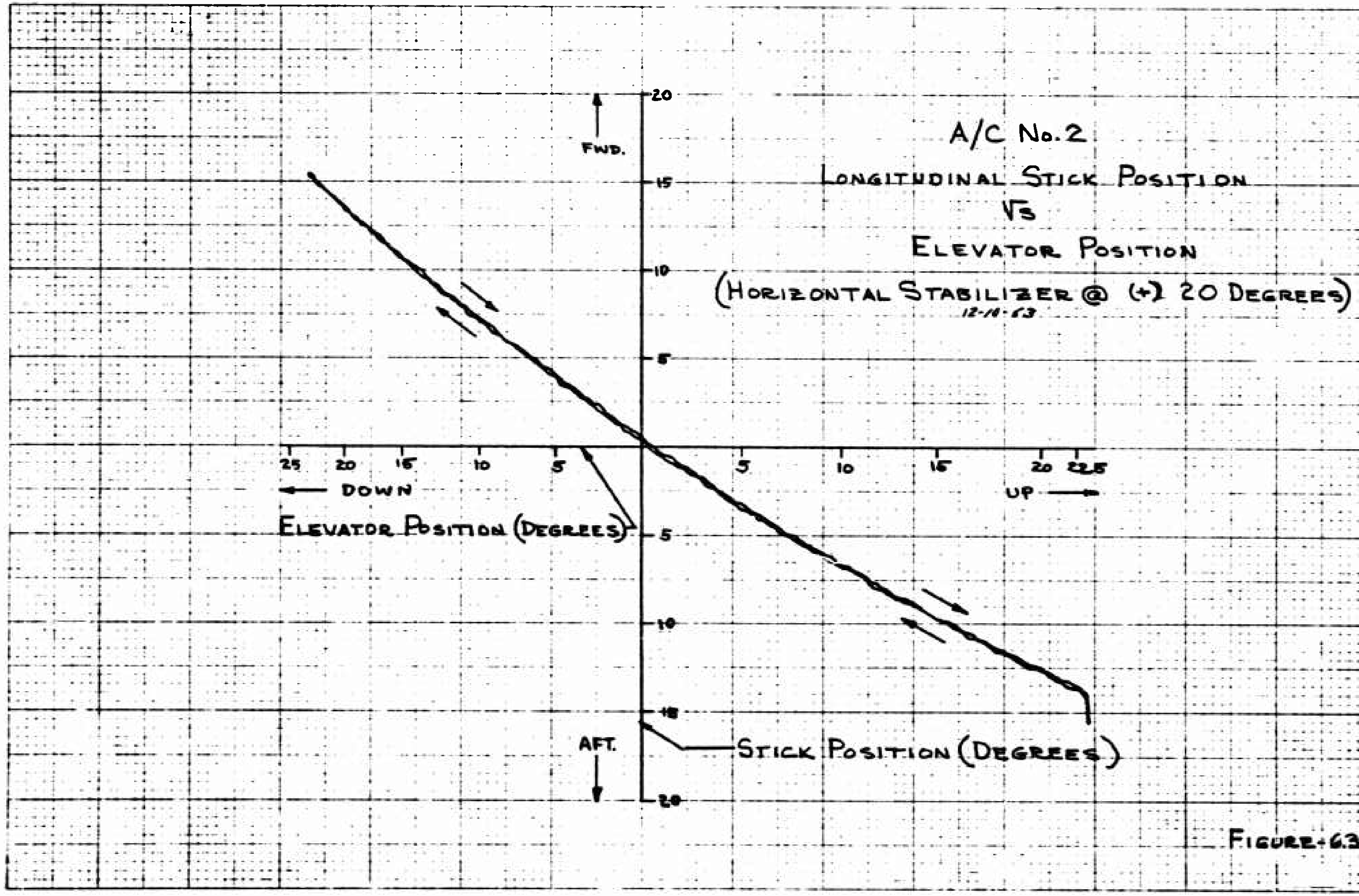

A/C No. 2
Longitudinal Stick Position
Vs
Elevator Position
(Horizontal Stabilizer @ (+) 20 Degrees)
12-11-63
FWD.
AFT.
20
15
10
5
5
10
15
20
25
20
15
10
5
5
10
15
20
22.5
DOWN
UP
Elevator Position (Degrees)
Stick Position (Degrees)

Figure 6.3.28

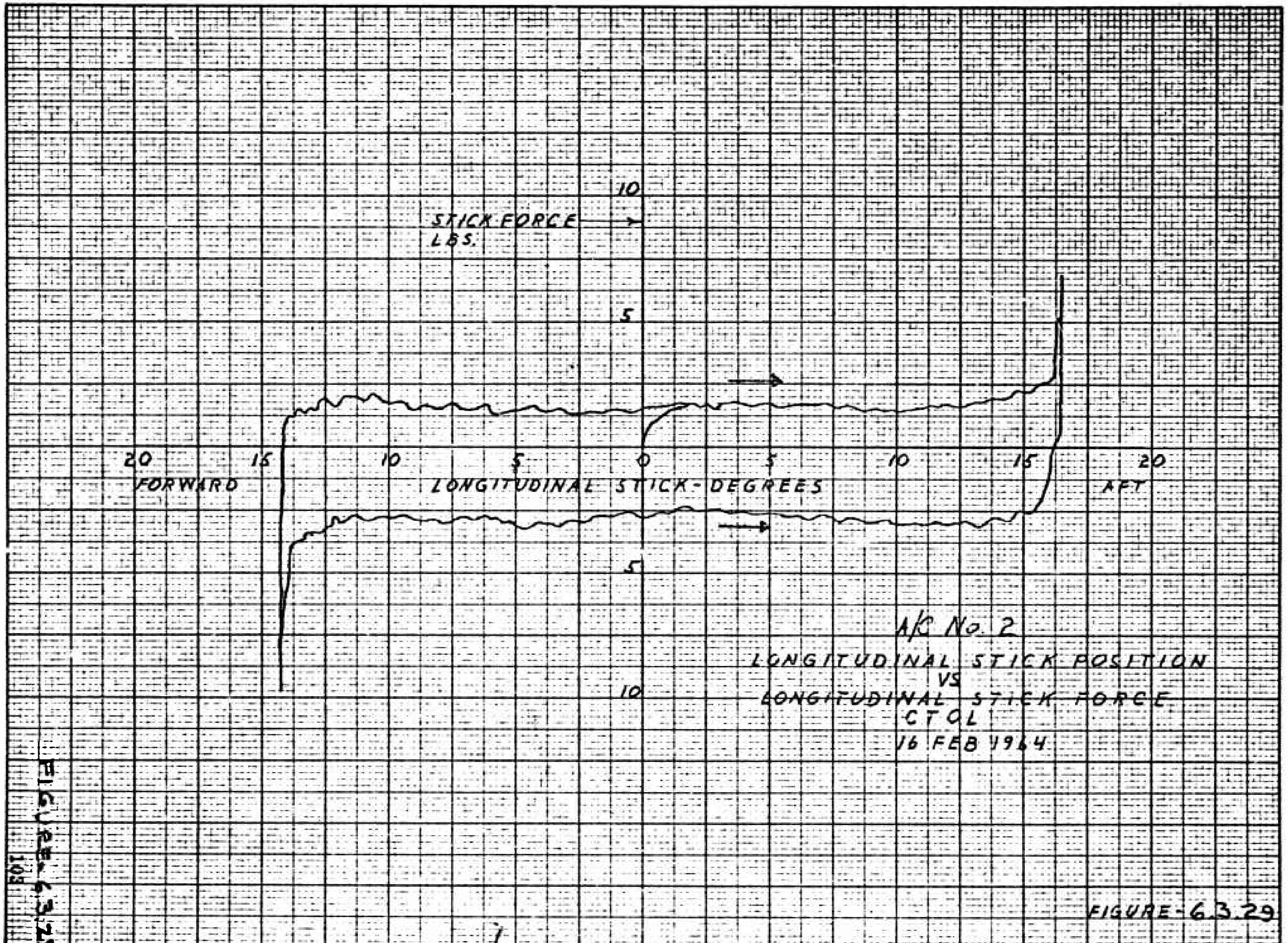

STICK FORCE LBS.
10
5
0
5
10
20
15
10
5
0
5
10
15
20
FORWARD
LONGITUDINAL STICK - DEGREES
AFT
A/C No. 2
LONGITUDINAL STICK POSITION
VS
LONGITUDINAL STICK FORCE
CTOL
16 FEB 1964

FIGURE - 6.3.29

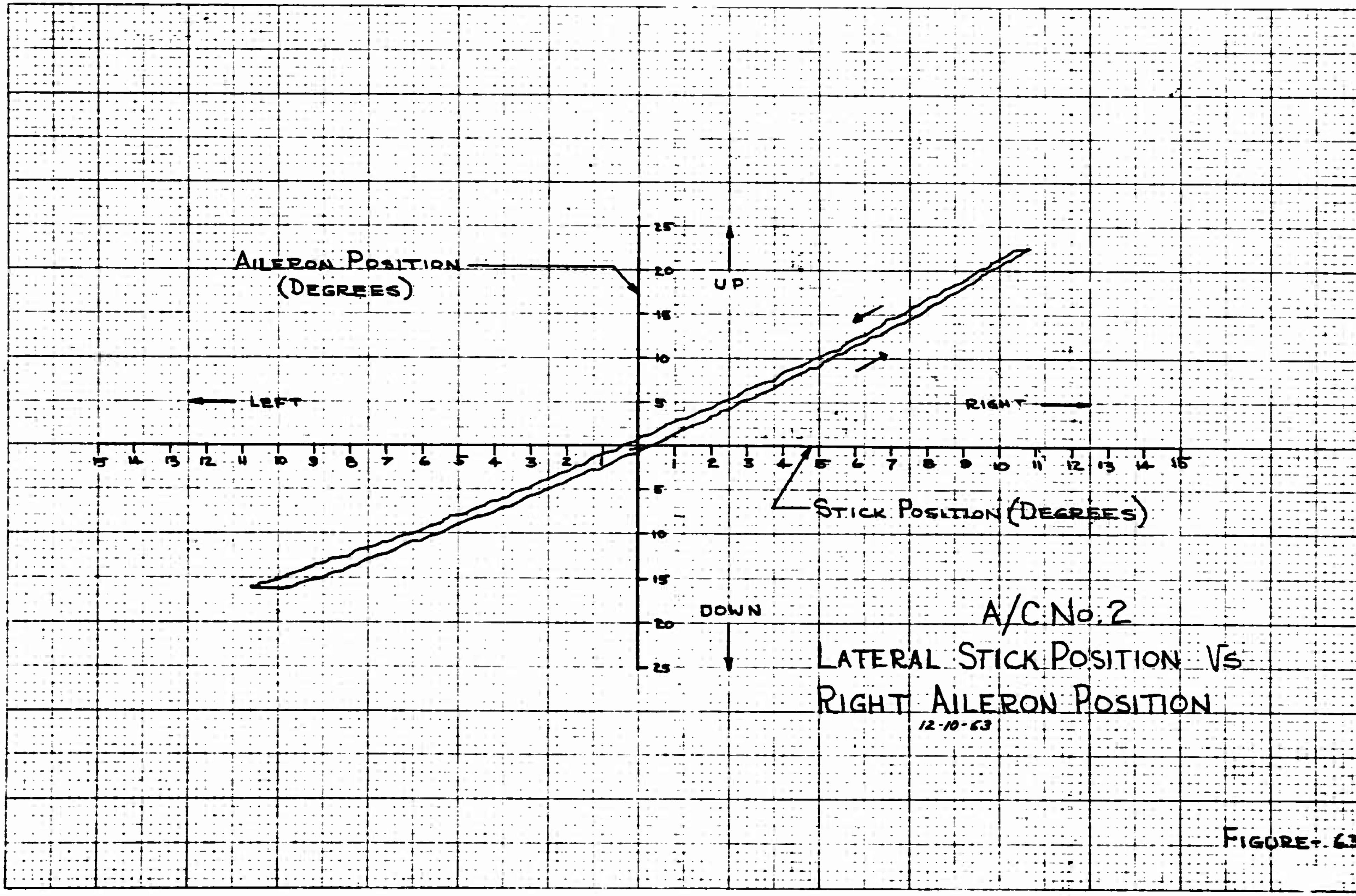

AILERON POSITION
(DEGREES)
UP
DOWN
LEFT
RIGHT
STICK POSITION (DEGREES)
25
20
15
10
5
5
10
15
20
25
15 14 13 12 11 10 9 8 7 6 5 4 3 2 1
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15
A/C No. 2
LATERAL STICK POSITION Vs
RIGHT AILERON POSITION
12-10-63

FIGURE- 6330

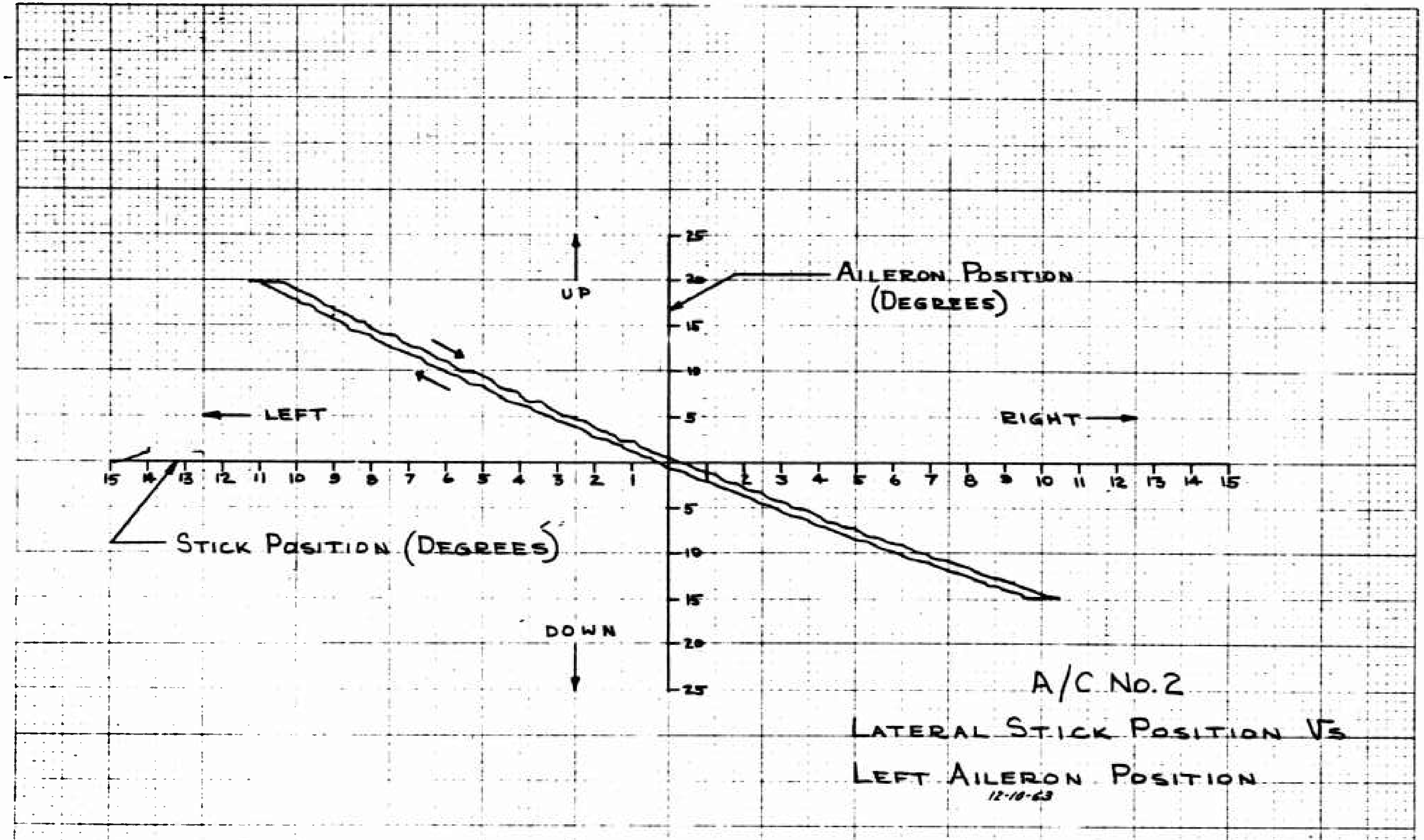
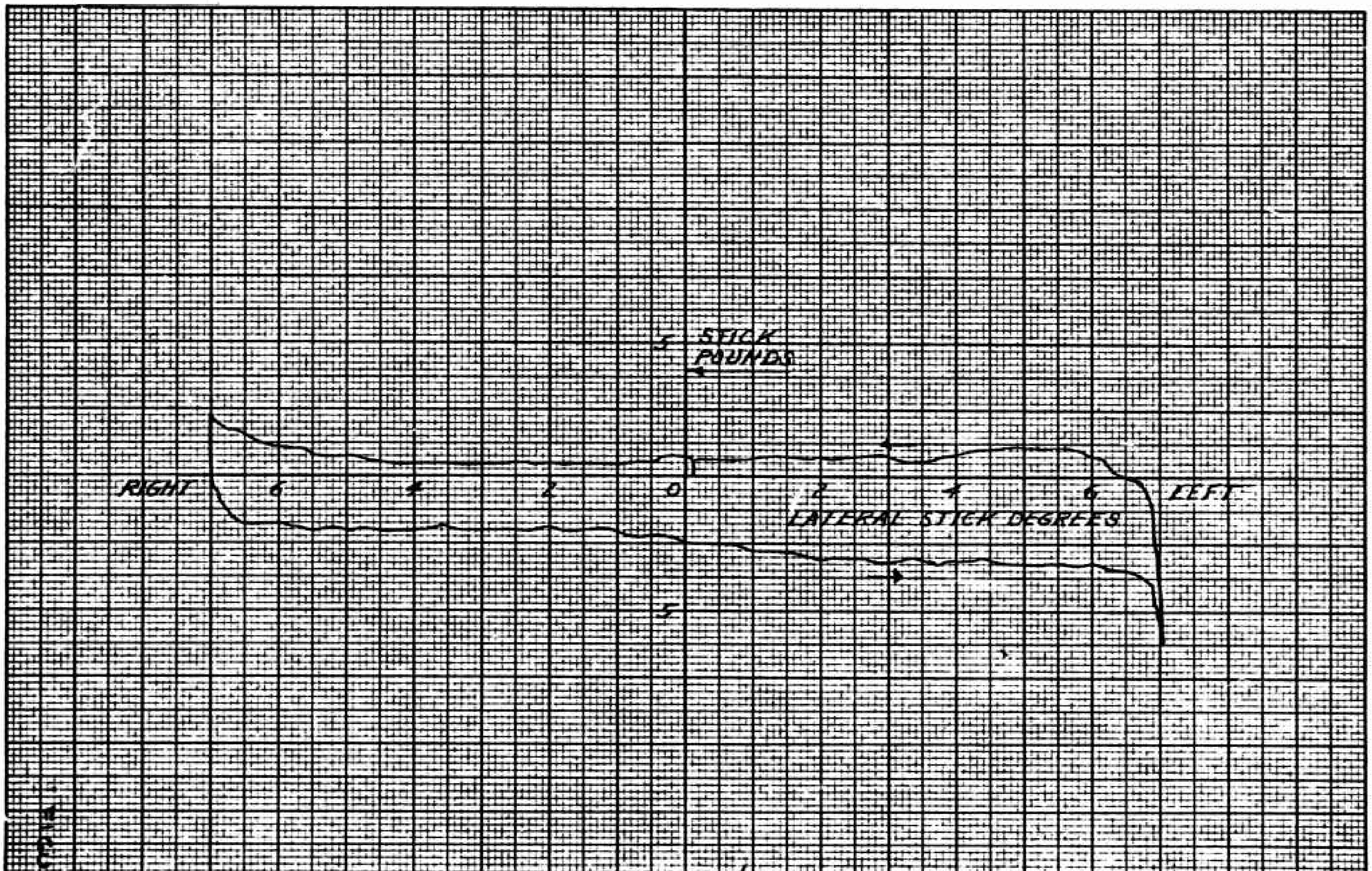
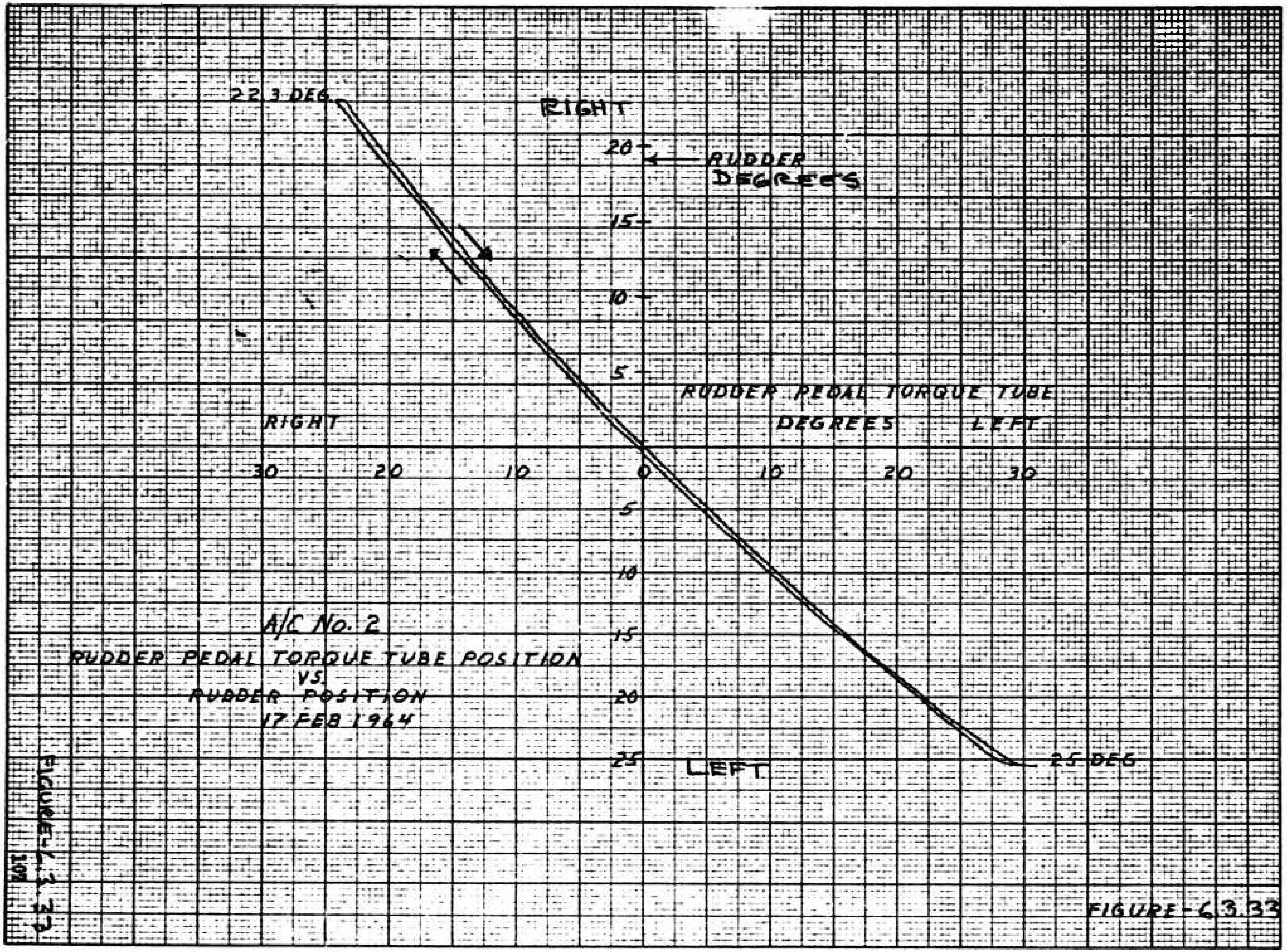

Aileron Position (Degrees)
Stick Position (Degrees)
Up
Down
Left
Right
A/C No. 2
Lateral Stick Position Vs
Left Aileron Position
12-10-63

Figure - 6.3.1



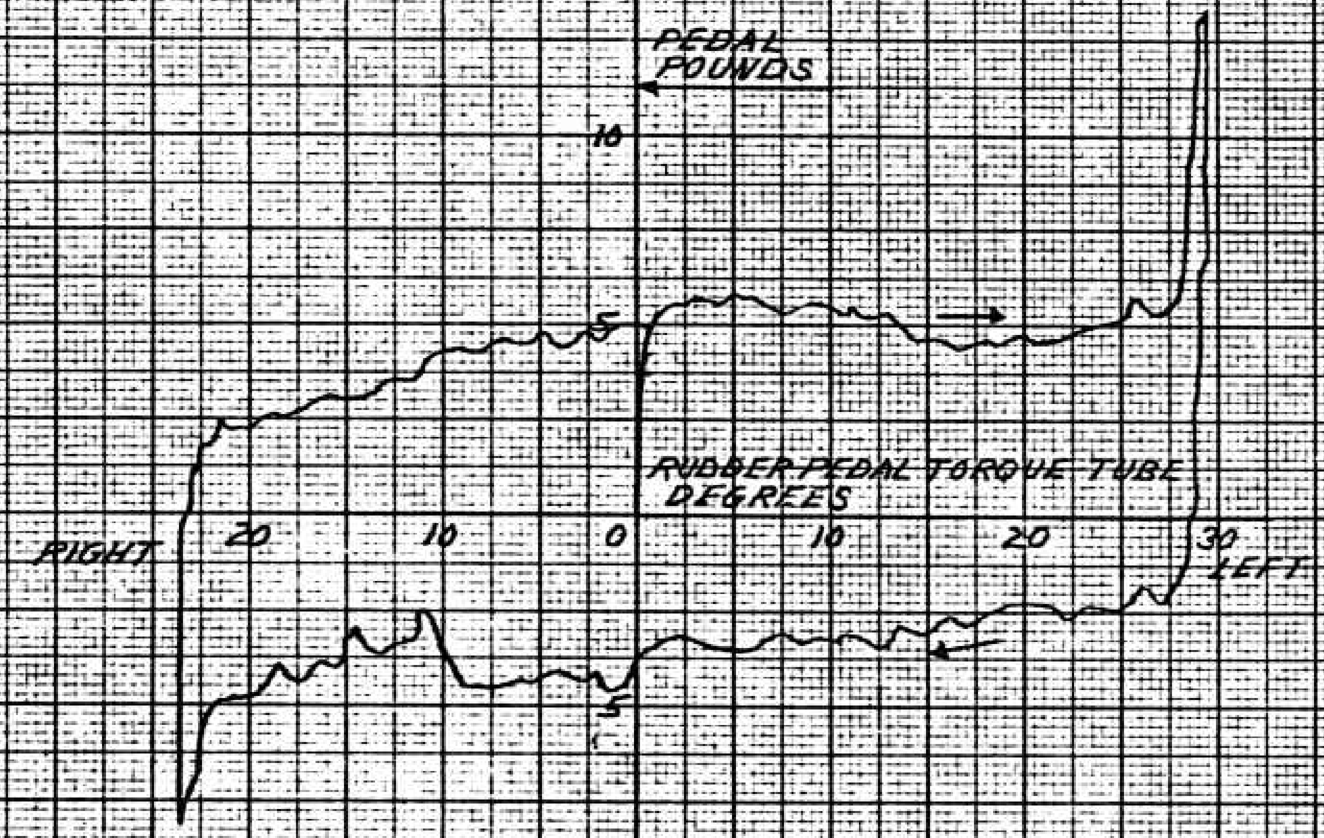
NO. 2 A/C
LATERAL STICK POSITION
VS.
LATERAL STICK FORCE
CTOL MODE, 18 FEB. 1964

FIGURE 6.3.32



22.3 DEG
RIGHT
RUDDER DEGREES
20
15
10
5
0
5
10
15
20
25
LEFT
RUDDER PEDAL TORQUE TUBE DEGREES
RIGHT
LEFT
30
20
10
0
10
20
30
25 DEG
A/C No. 2
RUDDER PEDAL TORQUE TUBE POSITION
VS.
RUDDER POSITION
17 FEB 1964
FIGURE-6.3.33

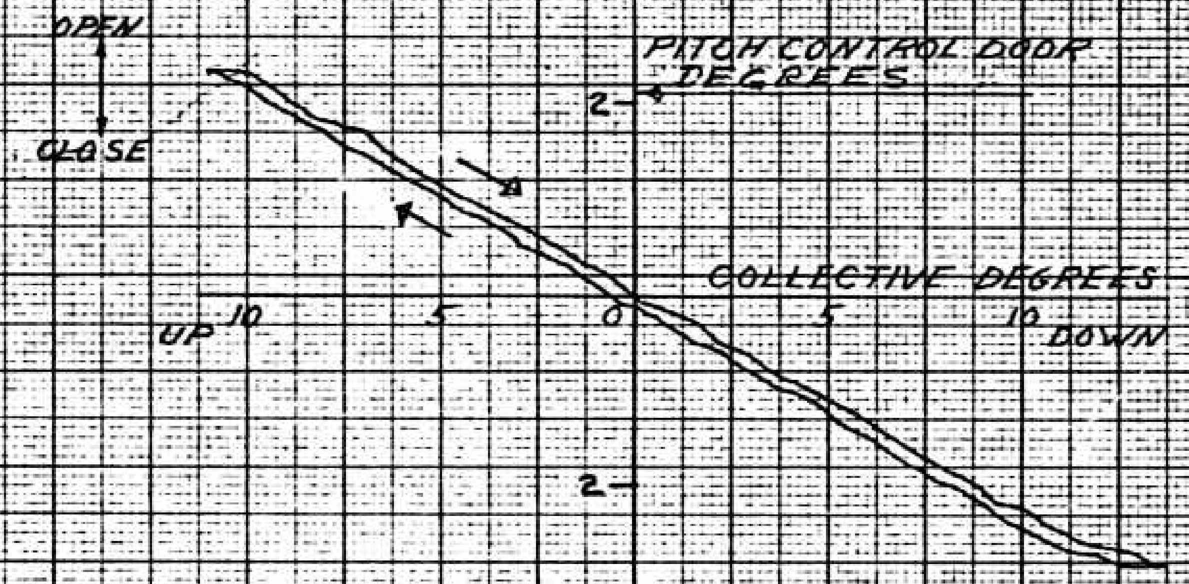
FIGURE-6.3.33



NO. 2 A/C
RUDDER PEDAL TORQUE TUBE POSITION
VS.
RUDDER PEDAL FORCE
CTOL MODE, 17 FEB 64

FIGURE 6.3.34

FIGURE-6.3.34



NO. 2 A/C
COLLECTIVE CONTROL POSITION
VS.
PITCH CONTROL DOOR POSITION
15 FEB. 64

FIGURE-6.3.35

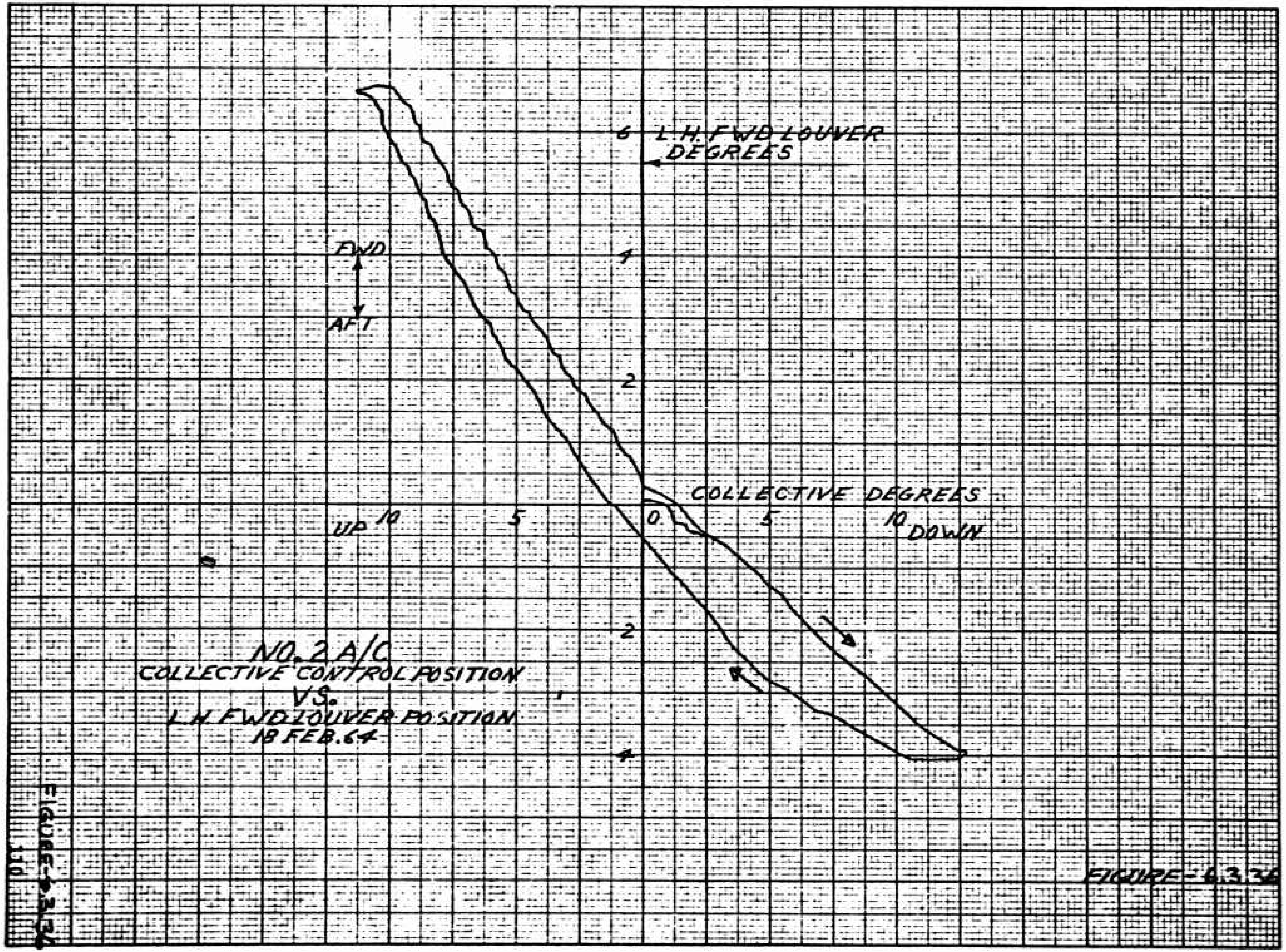

L.H. FWD LOUVER DEGREES
6
4
2
2
4
FWD
AFT
COLLECTIVE DEGREES
UP 10
5
0
5
10 DOWN
NO. 2 A/C
COLLECTIVE CONTROL POSITION
VS.
L.H. FWD LOUVER POSITION
18 FEB. 64

FIGURE-6.3.36

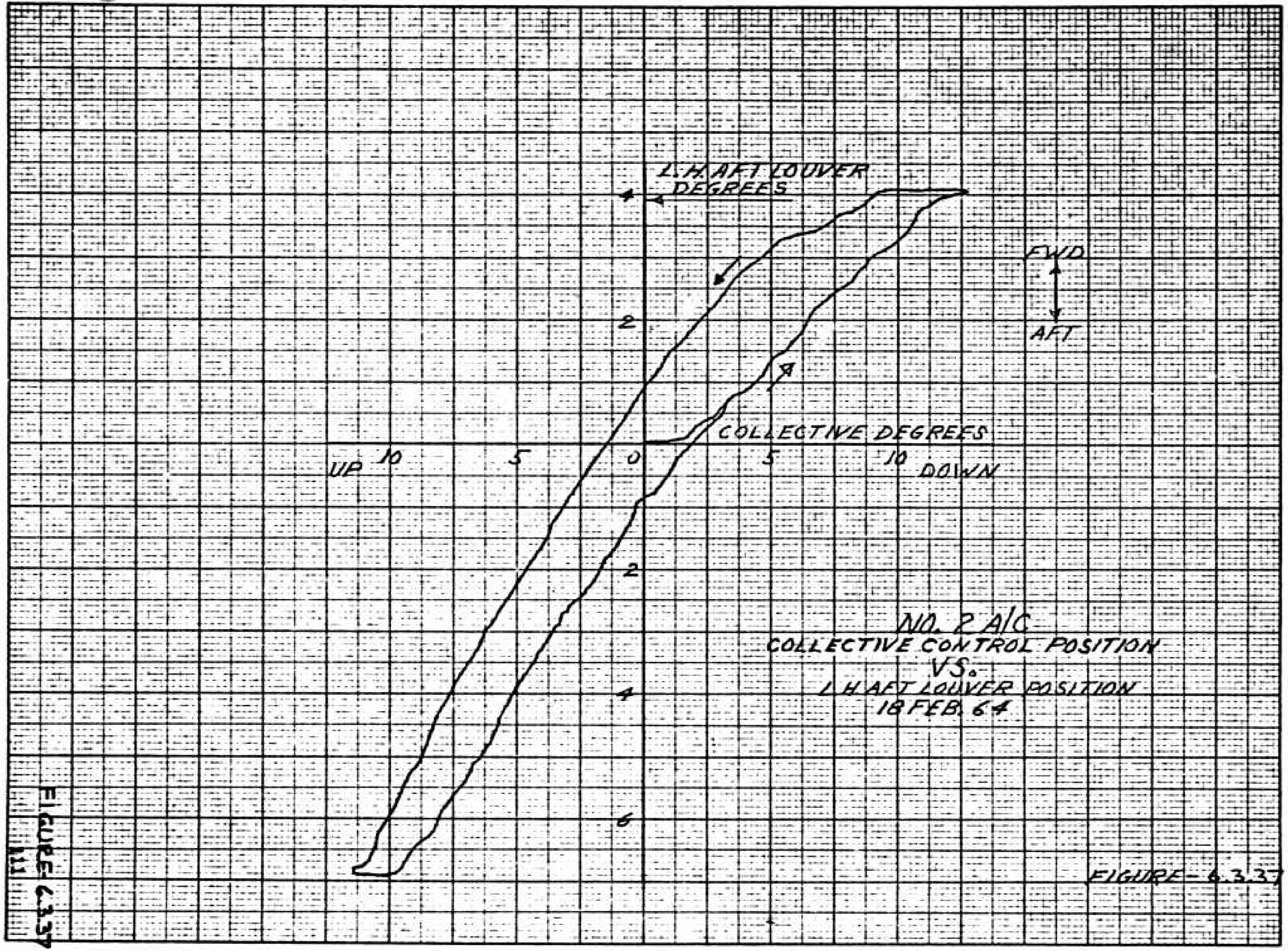


FIGURE-6.3.37

FIGURE 6.3.37

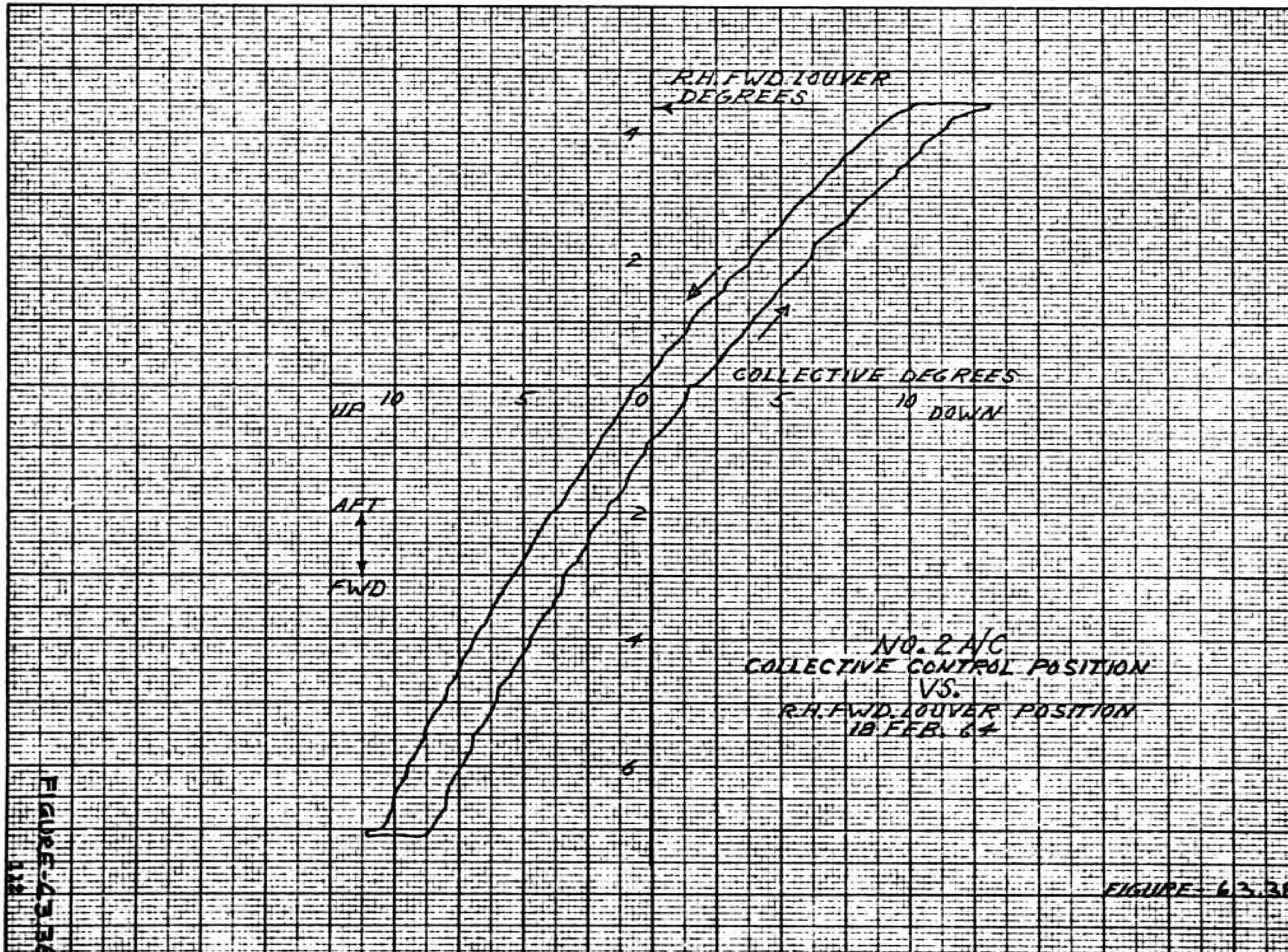


FIGURE-63.38

FIGURE-63.38

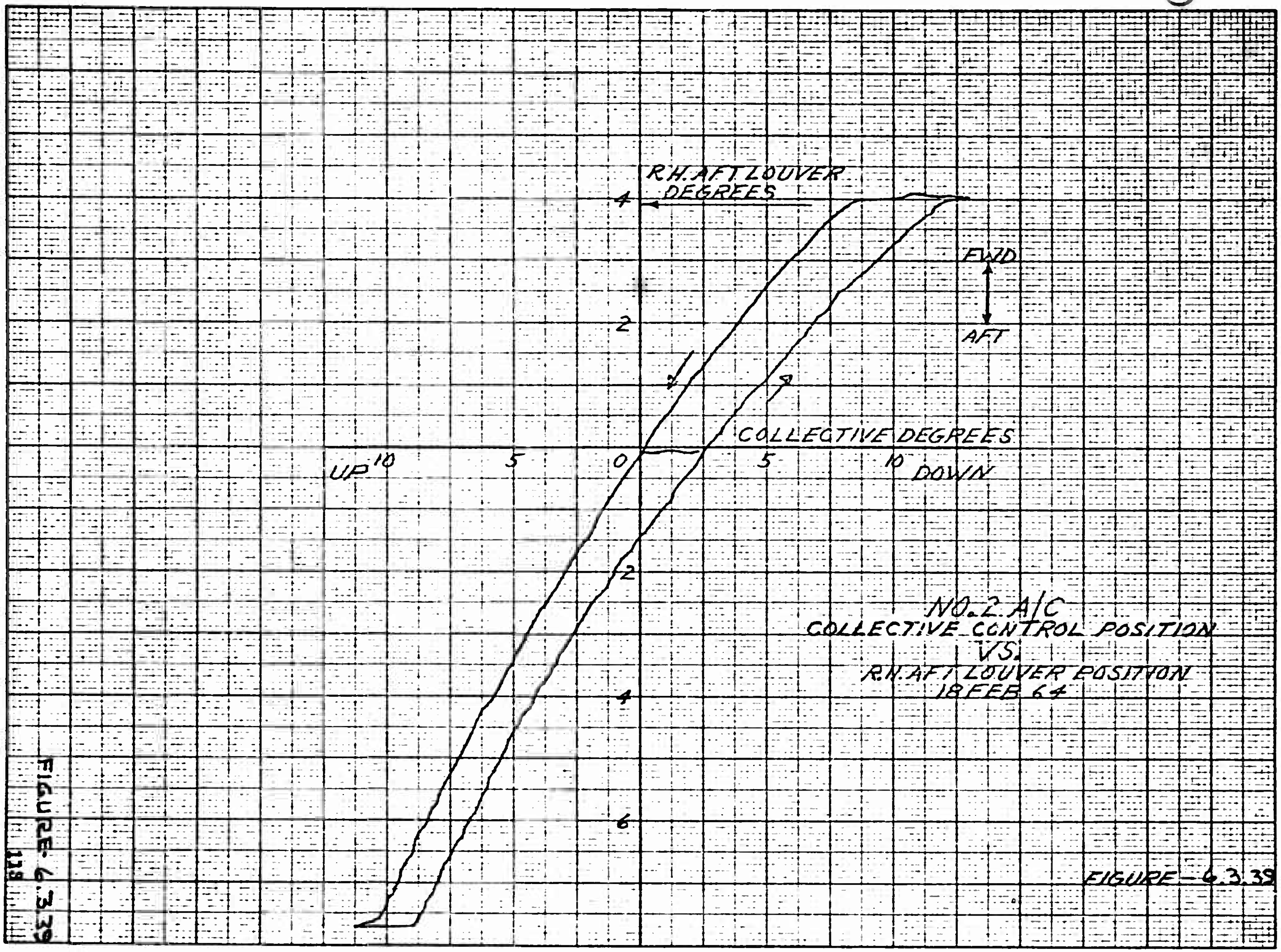
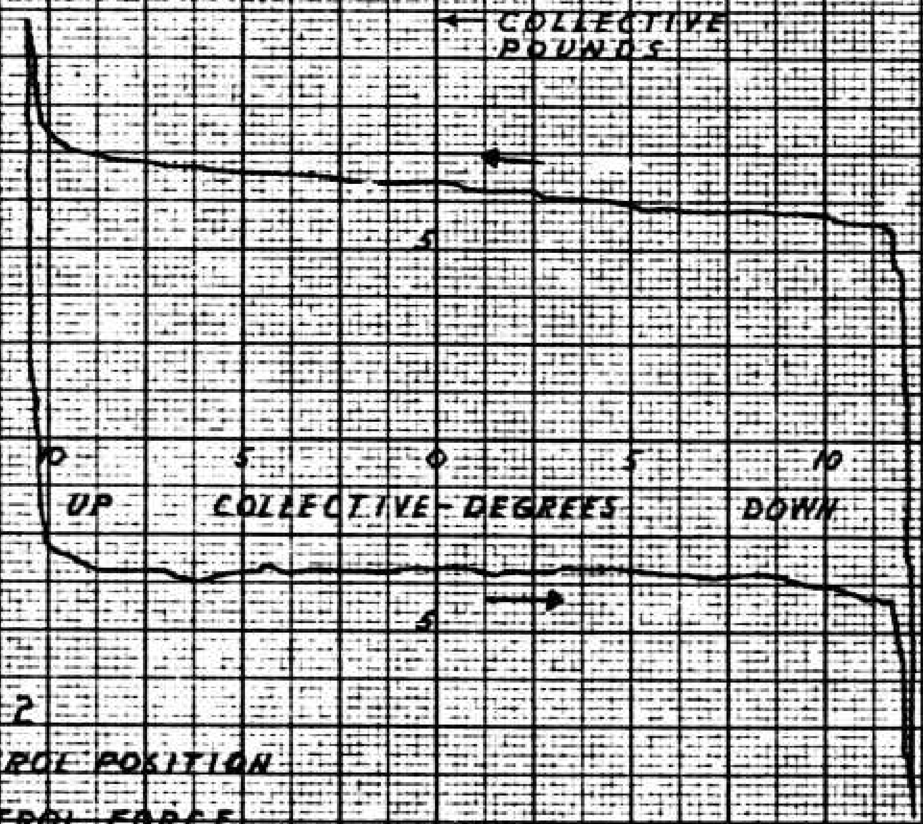


FIGURE - 6.3.39

FIGURE 6.3.39



A/C No. 2

COLLECTIVE CONTROL POSITION
VS.
COLLECTIVE CONTROL FORCE
15 FEB 1964

FIGURE - 6.3.40

FIGURE - 6.3.40

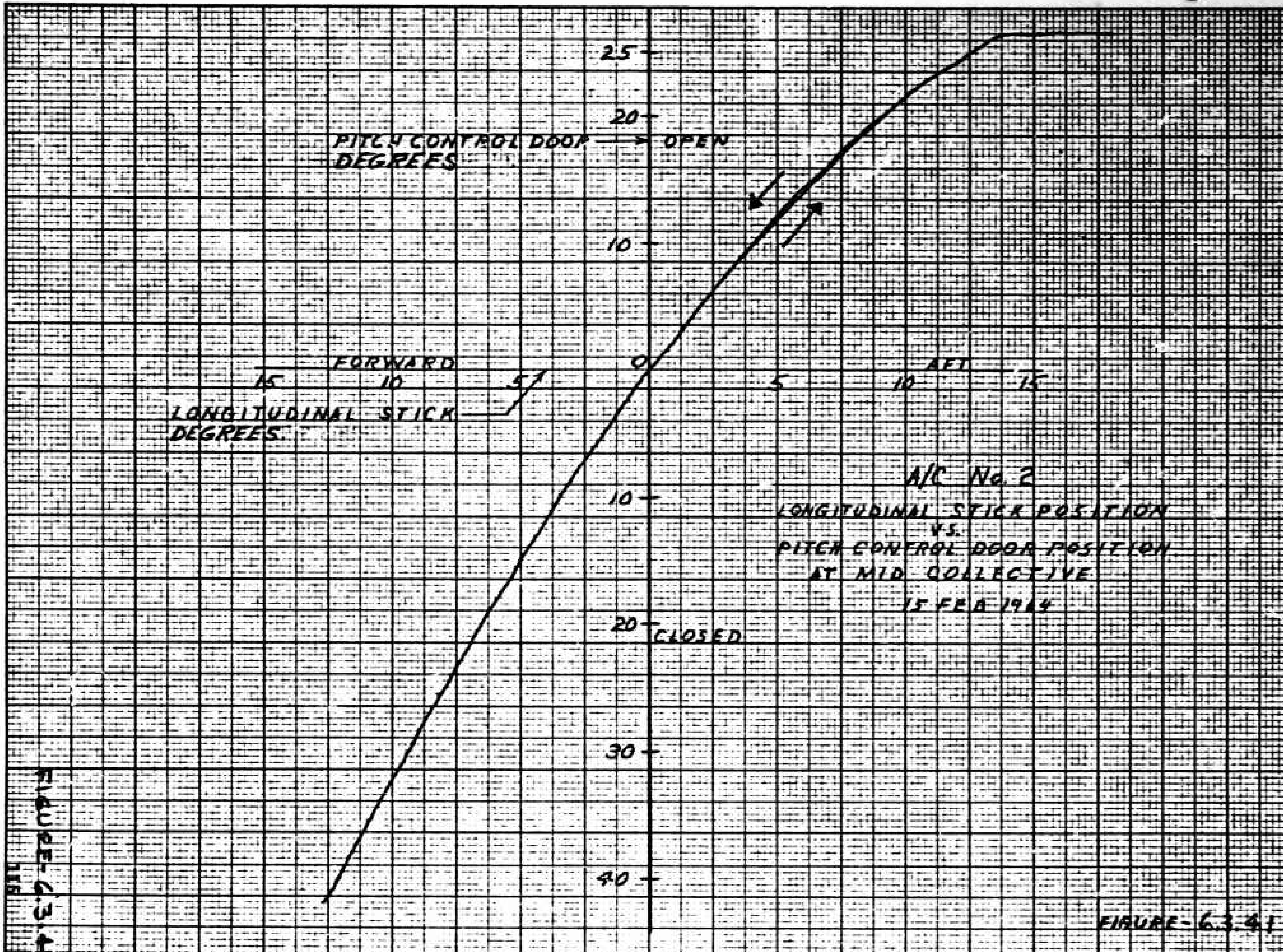

A/C No. 2
LONGITUDINAL STICK POSITION
VS.
PITCH CONTROL DOOR POSITION
AT MID COLLECTIVE
15 FEB 1964
PITCH CONTROL DOOR DEGREES
OPEN
CLOSED
LONGITUDINAL STICK DEGREES
FORWARD
AFT
25
20
10
0
10
20
30
40
15
10
5
5
10
15
FIGURE - 6.3.41

FIGURE-6.3.41

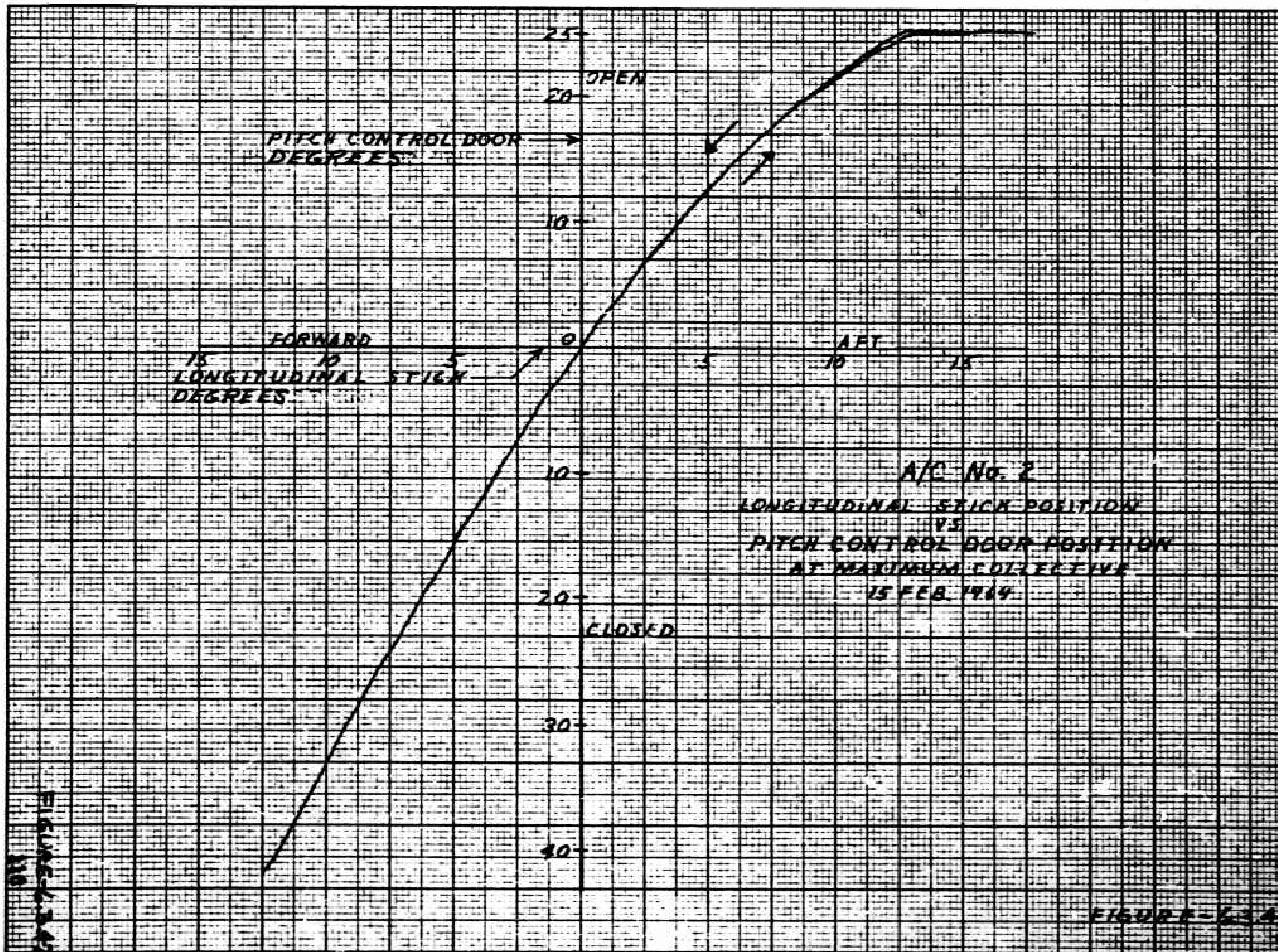


FIGURE-6.3.42

FIGURE-6.3.42

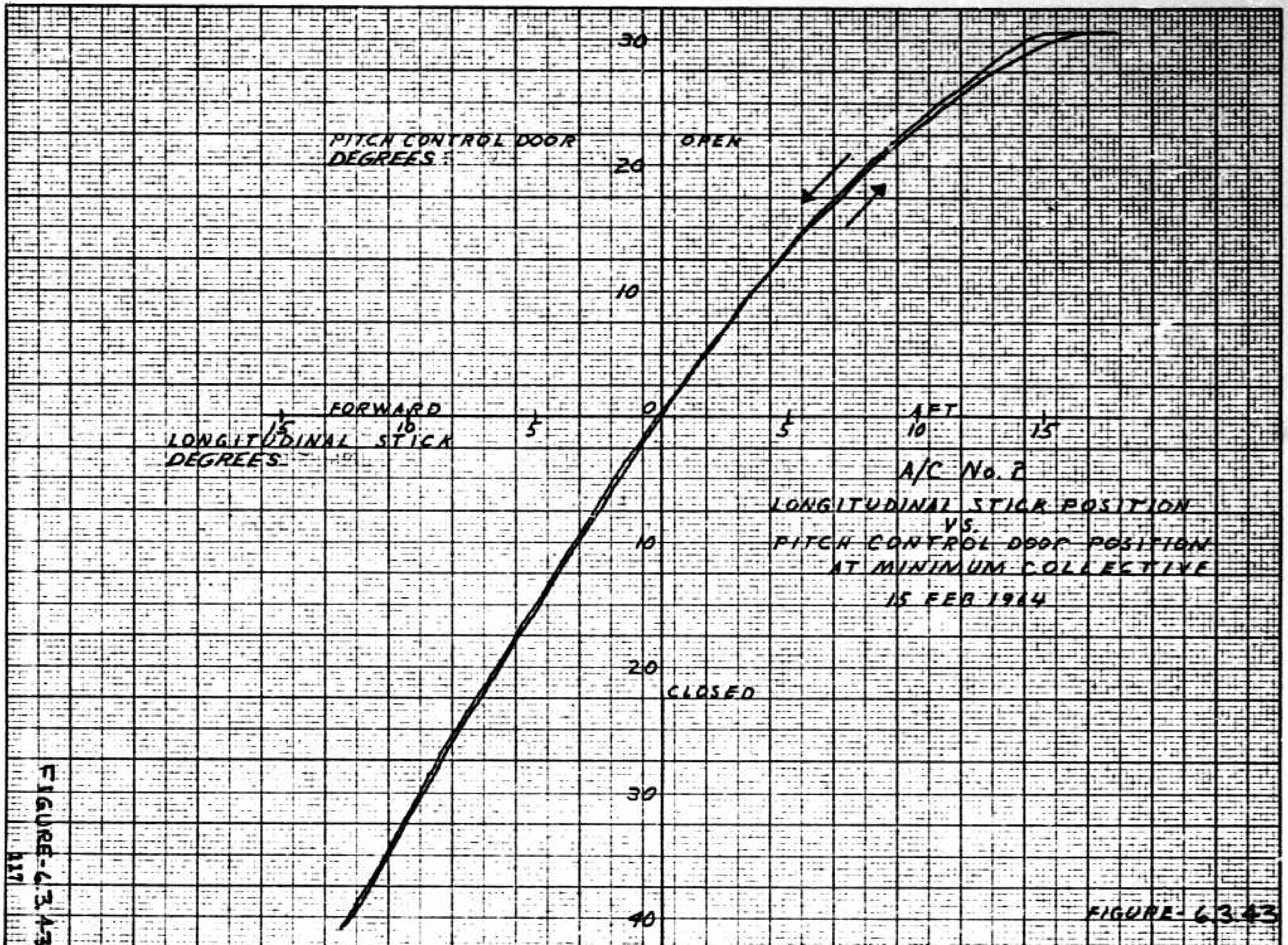


FIGURE-6.3.43

FIGURE-6.3.43

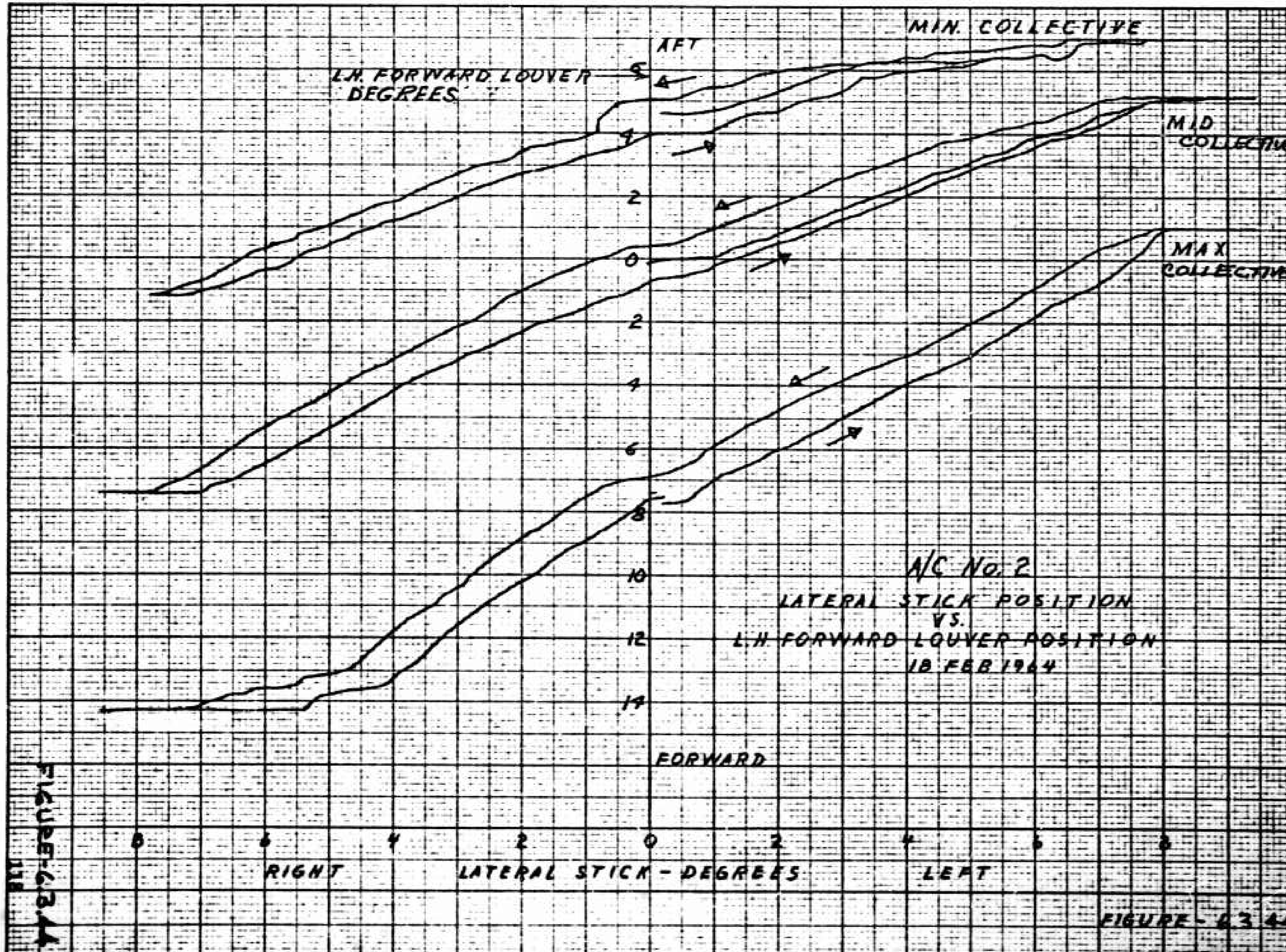


FIGURE-6.3.44

FIGURE-6.3.44

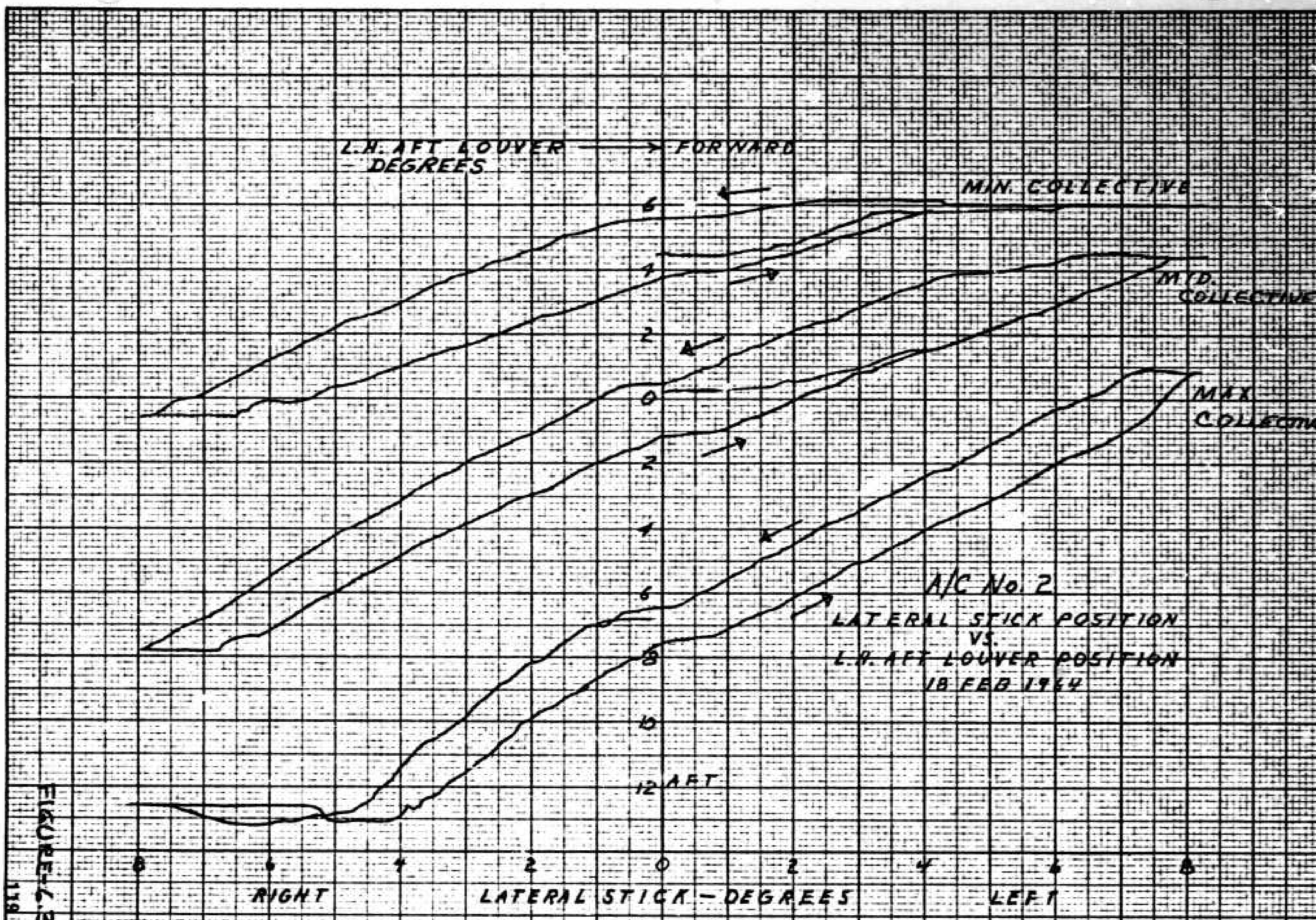
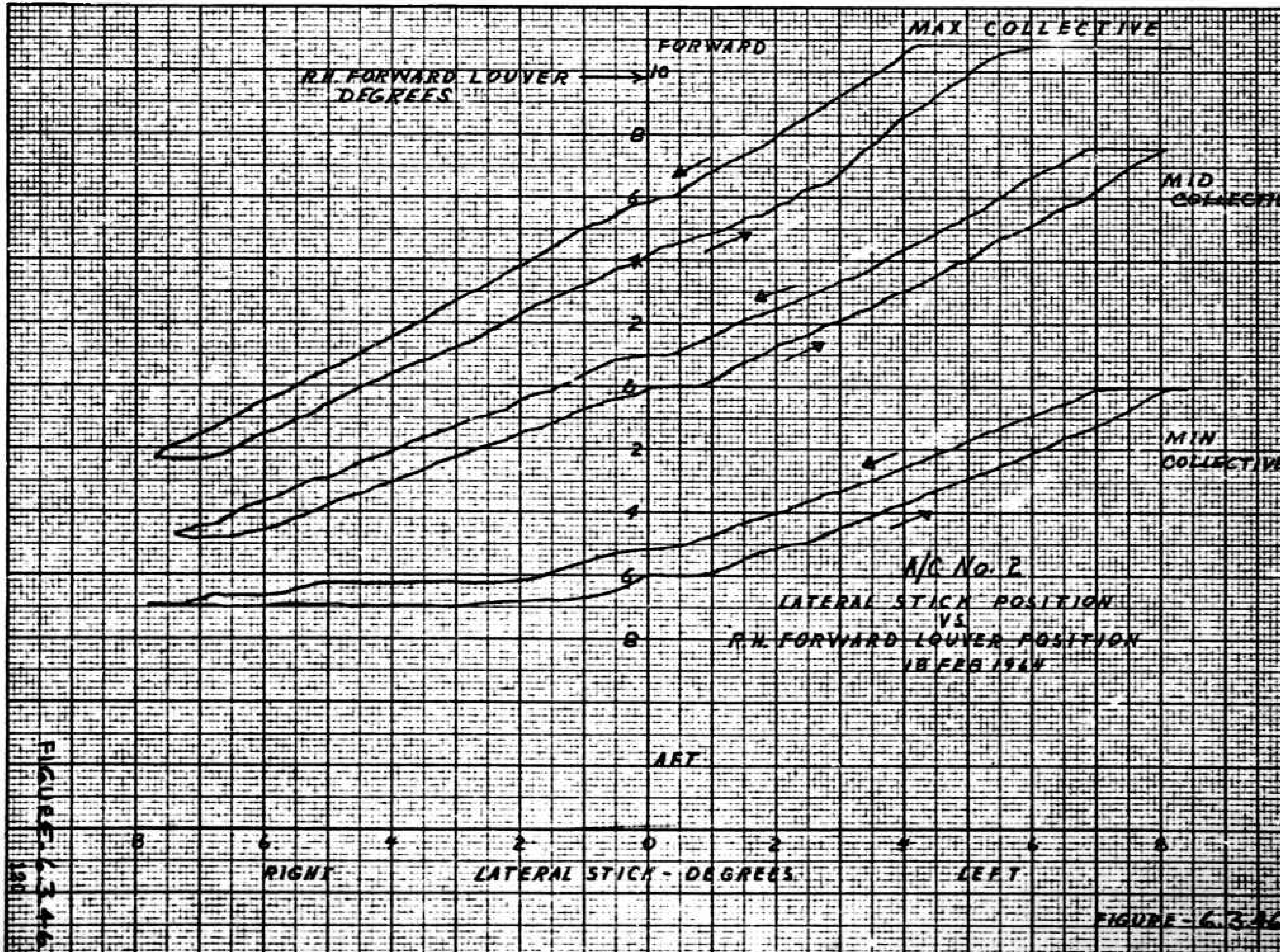


FIGURE-6.3.45

FIGURE-6.3.45



MAX COLLECTIVE
FORWARD
R.H. FORWARD LOUVER DEGREES
10
8
6
4
2
0
2
4
6
8
MID COLLECTIVE
MIN COLLECTIVE
A/C No. 2
LATERAL STICK POSITION VS R.H. FORWARD LOUVER POSITION
18 FEB 1964
AFT
8
6
4
2
0
2
4
6
8
RIGHT
LATERAL STICK - DEGREES
LEFT
FIGURE - 6.3.46

FIGURE 6.3.46

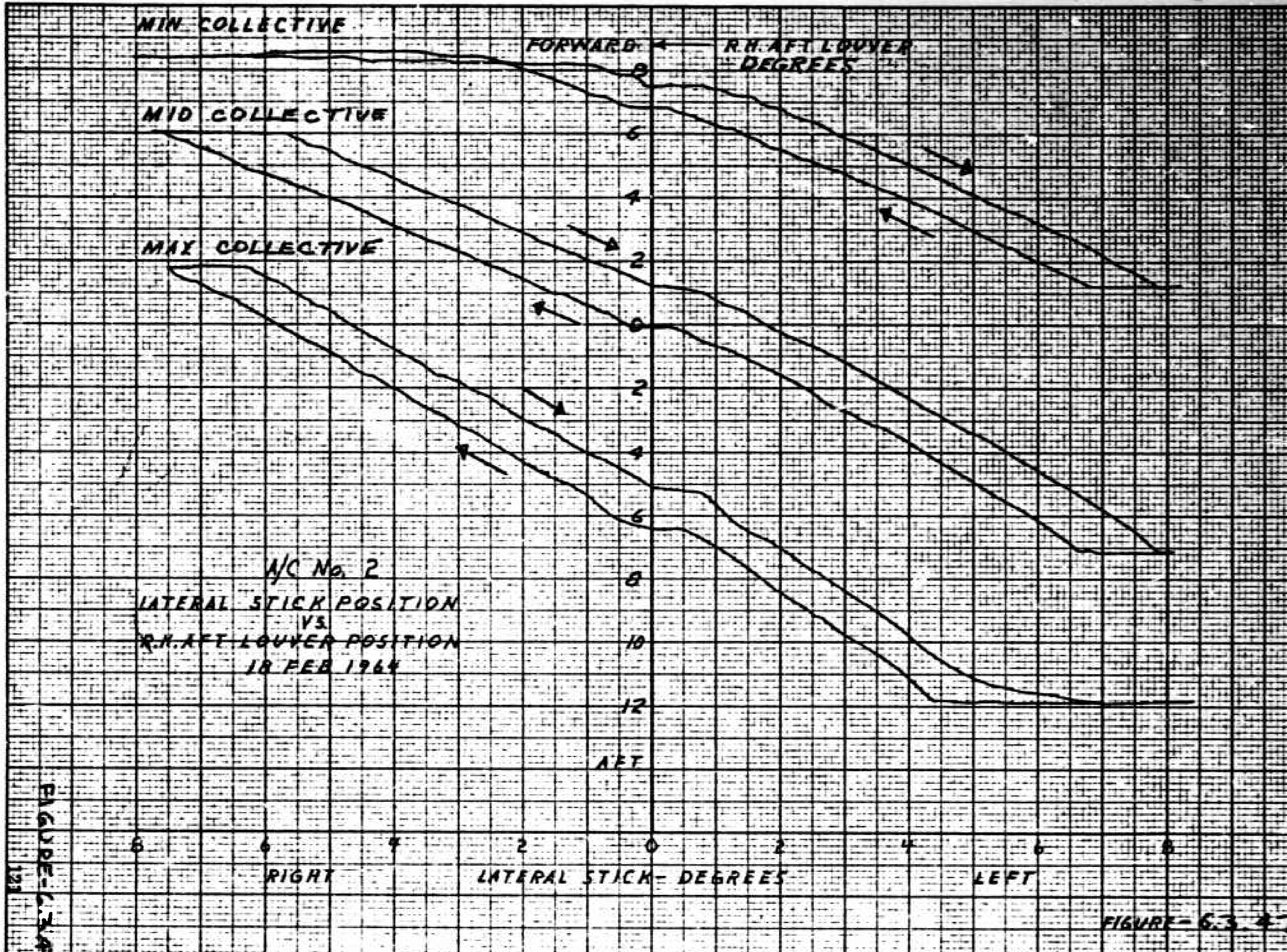
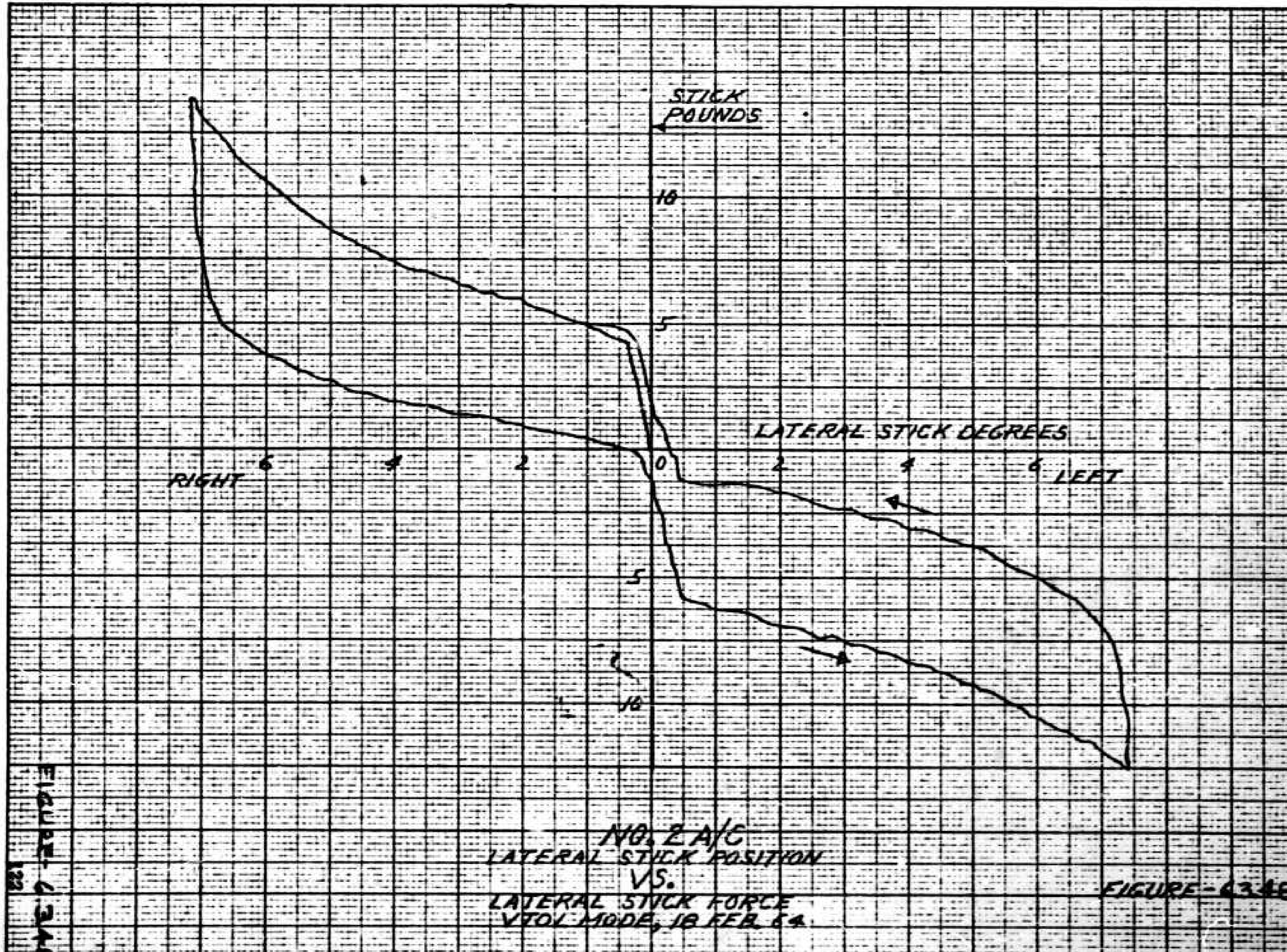
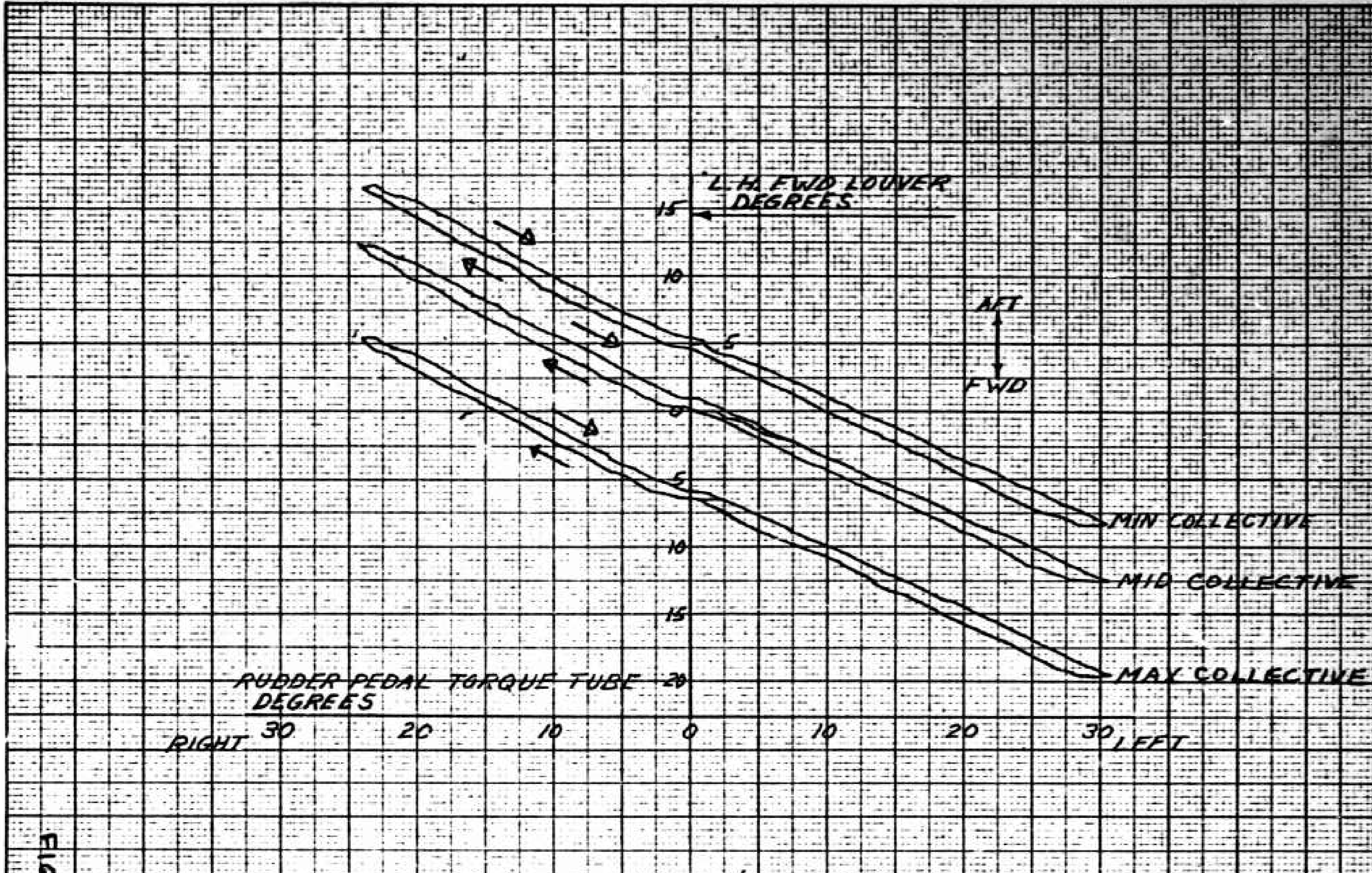

MIN COLLECTIVE
MID COLLECTIVE
MAX COLLECTIVE
FORWARD ← R.H. AFT LOUVER DEGREES
8
6
4
2
0
2
4
6
8
10
12
AFT
A/C No. 2
LATERAL STICK POSITION
VS.
R.H. AFT LOUVER POSITION
18 FEB 1964
8
6
4
2
0
2
4
6
8
RIGHT
LATERAL STICK - DEGREES
LEFT
FIGURE-6.3.47

FIGURE-6.3.47



NO. 2 A/C
LATERAL STICK POSITION
VS.
LATERAL STICK FORCE
VTOL MODE, 18 FEB 64

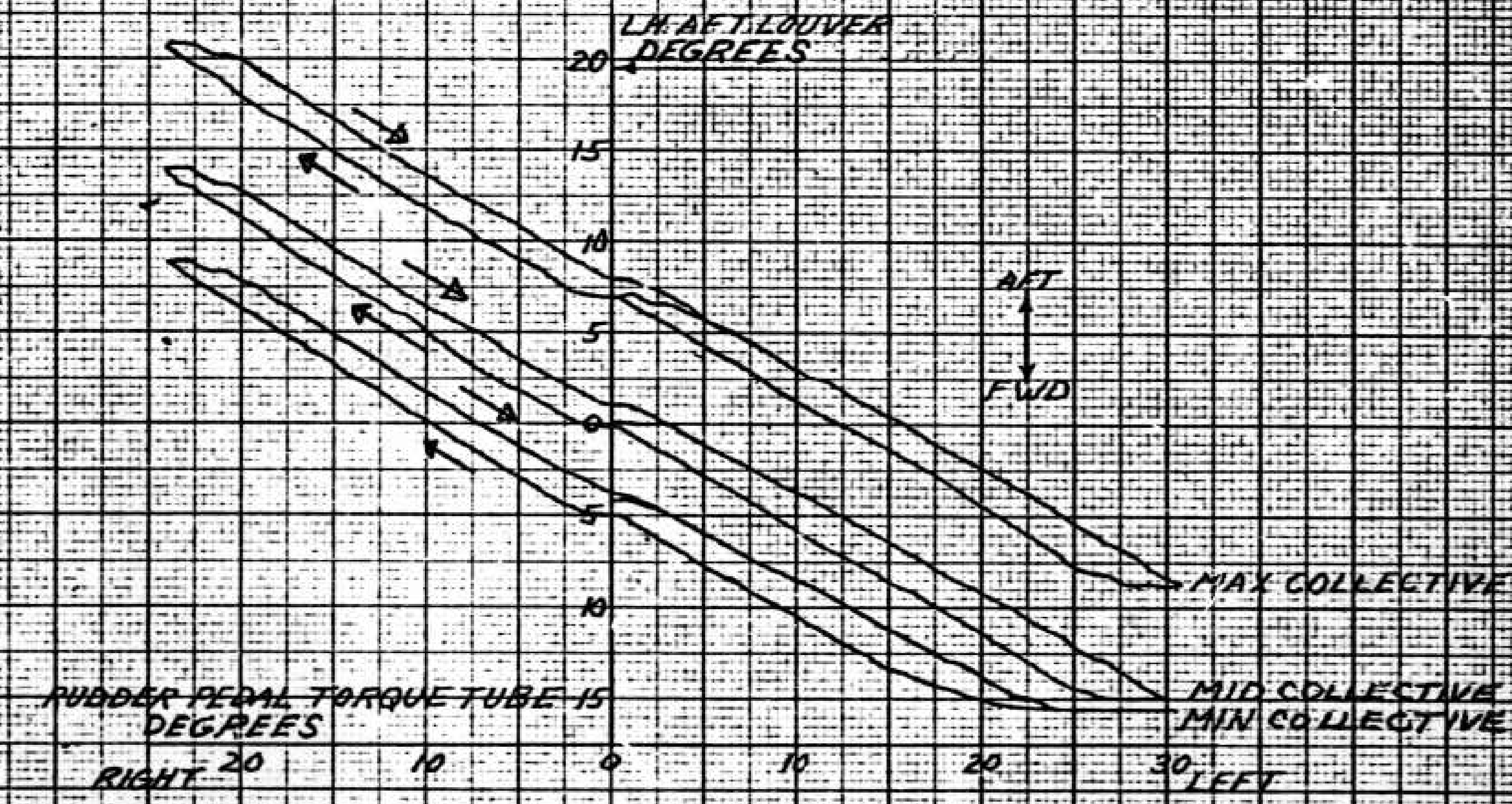
FIGURE-6.34B



NO. 2 A/C
RUDDER PEDAL TORQUE TUBE POSITION
VS.
L.H. FWD LOUVER POSITION
16 FEB 64

FIGURE-6.3.49

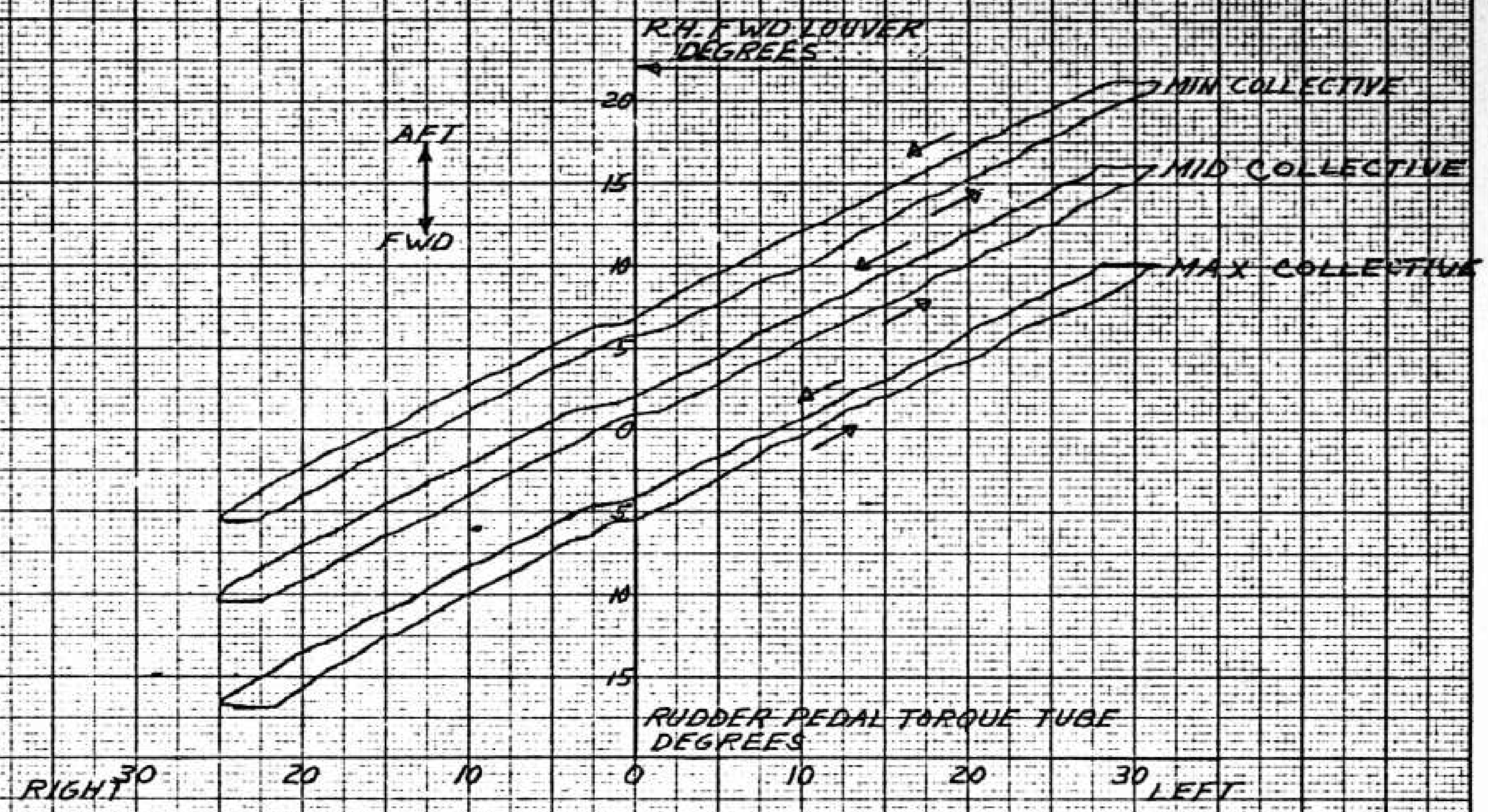
FIGURE-6.3.49



N.O. 2 A/C
RUDDER PEDAL TORQUE TUBE POSITION
VS.
L.H. AFT LOUVER POSITION
16 FEB. 64

FIGURE-6.3.50

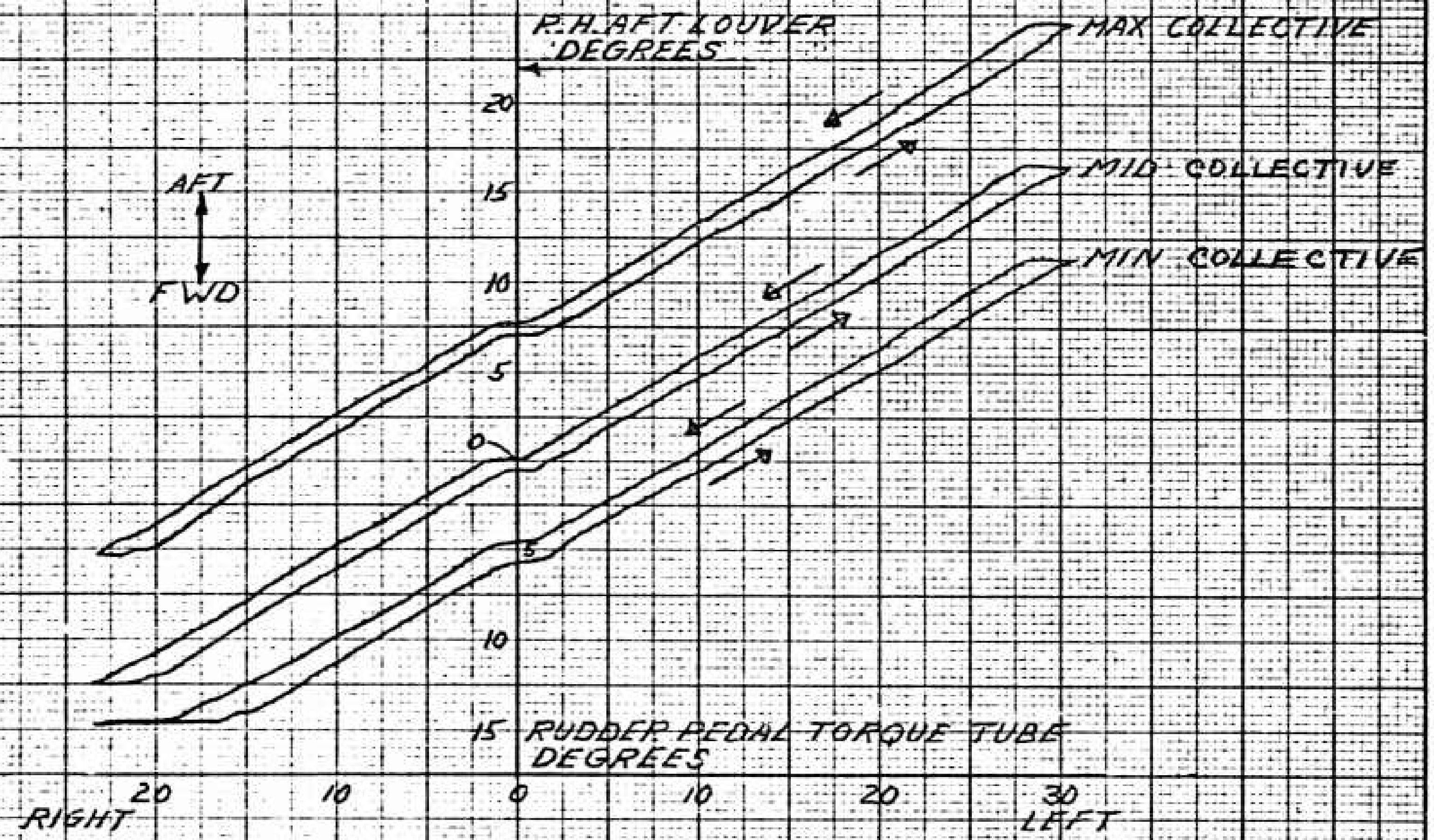
FIGURE-6.3.50



NO. 2 A/C
RUDDER PEDAL TORQUE TUBE POSITION
VS.
R.H. FWD LOUVER POSITION
16 FEB 64

FIGURE-6.3.51

FIGURE-6.3.51



NO. 2 A/C
RUDDER PEDAL TORQUE TUBE POSITION
VS.
R.H. AFT LOUVER POSITION
16 FEB. 64

FIGURE-6.3.52

FIGURE-6.3.52

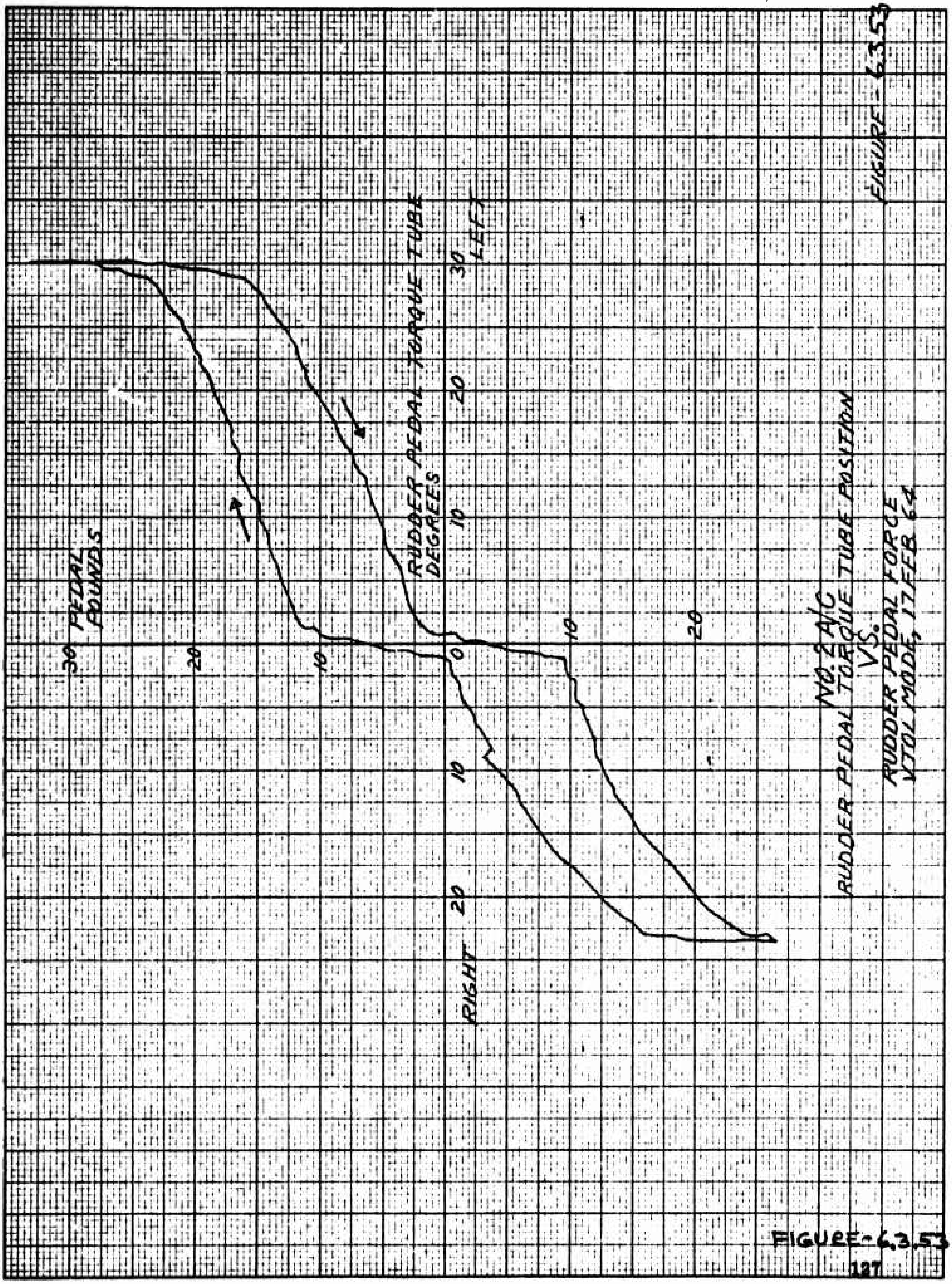

PEDAL POUNDS
RUDDER PEDAL TORQUE TUBE DEGREES
LEFT
RIGHT
30
20
10
0
10
20
30
NO.2 A/C
RUDDER PEDAL TORQUE TUBE POSITION
VS.
RUDDER PEDAL FORCE
VTOL MODE, 17 FEB 64
FIGURE-6.3.53

FIGURE-6.3.53

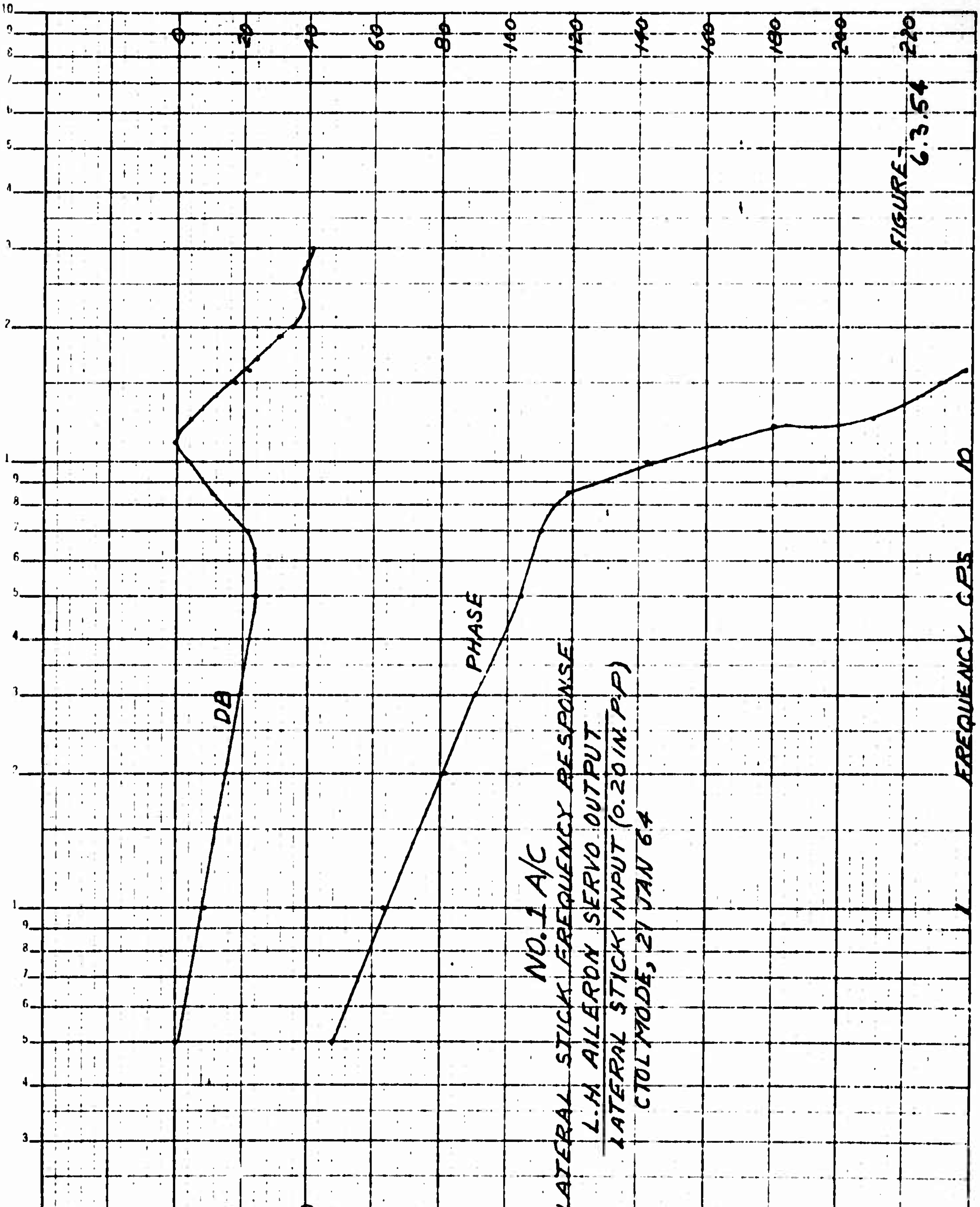


FIGURE- 6.3.54
NO. 1 A/C
LATERAL STICK FREQUENCY RESPONSE
L.H. AILERON SERVO OUTPUT
LATERAL STICK INPUT (0.20 IN. P.P)
CTOL MODE, 21 JAN 64
PHASE
DB
DB
FREQUENCY CPS
0
20
40
60
80
100
120
140
160
180
200
220
0
10
20
1
10

FIGURE- 6.3.54

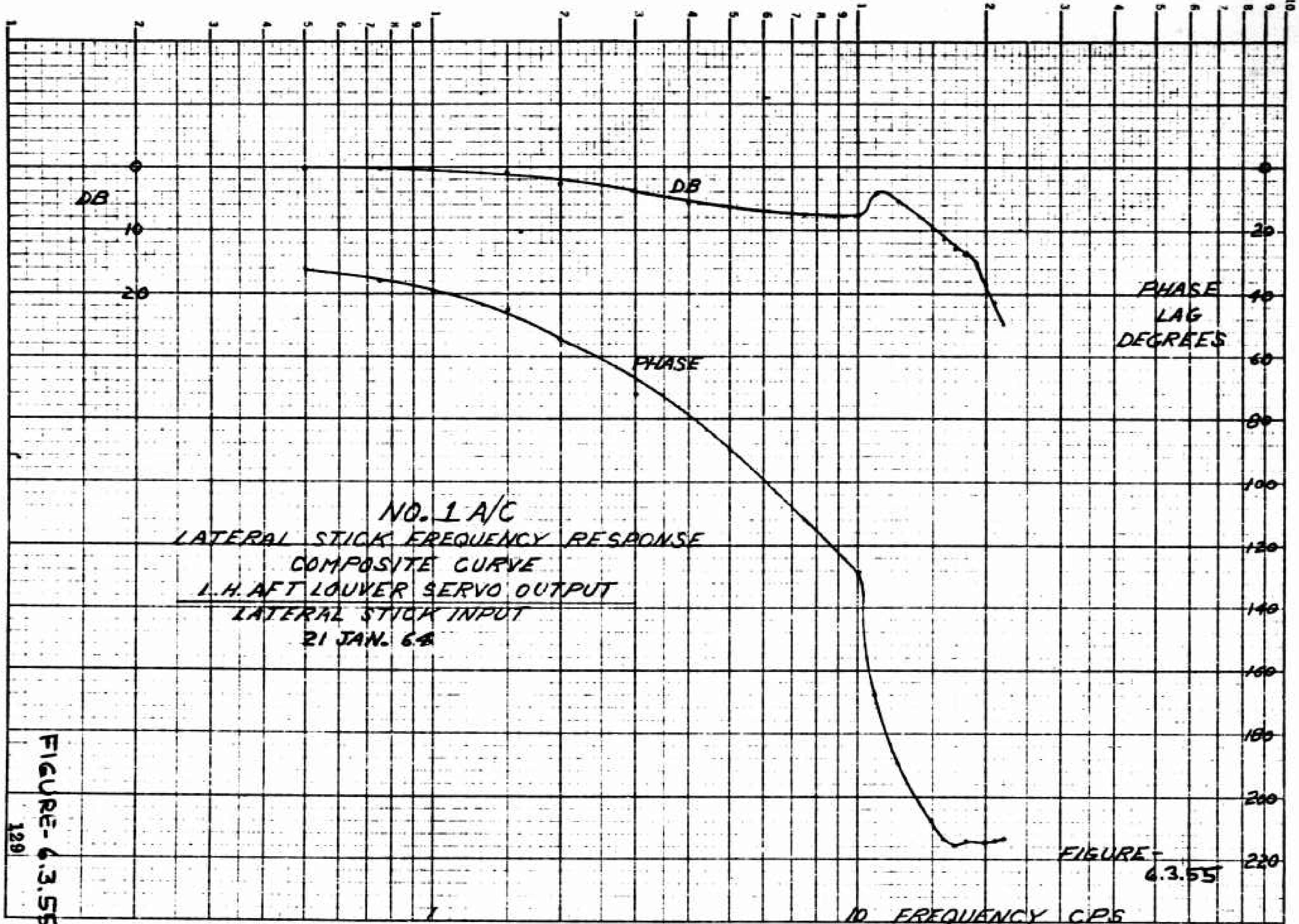
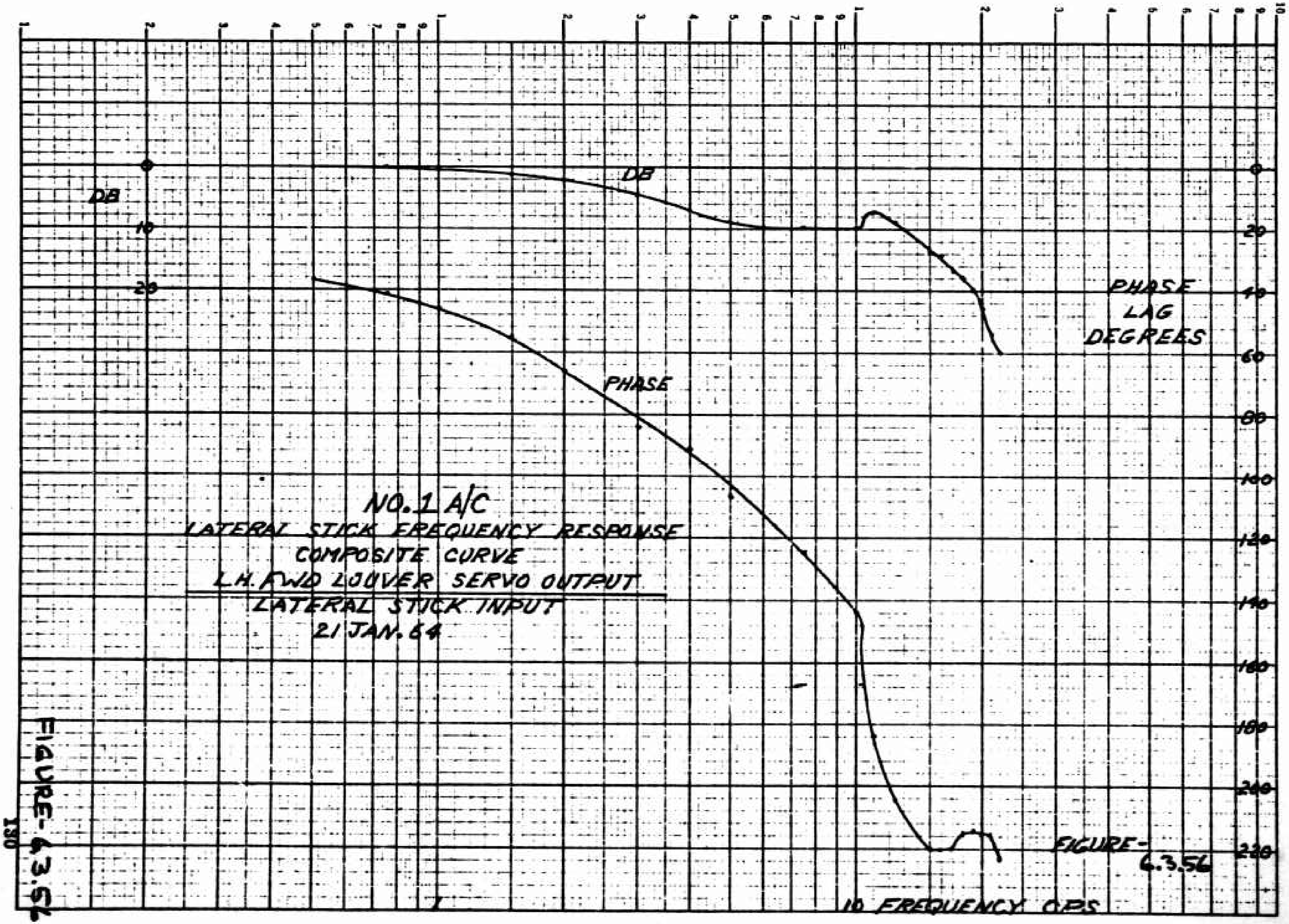
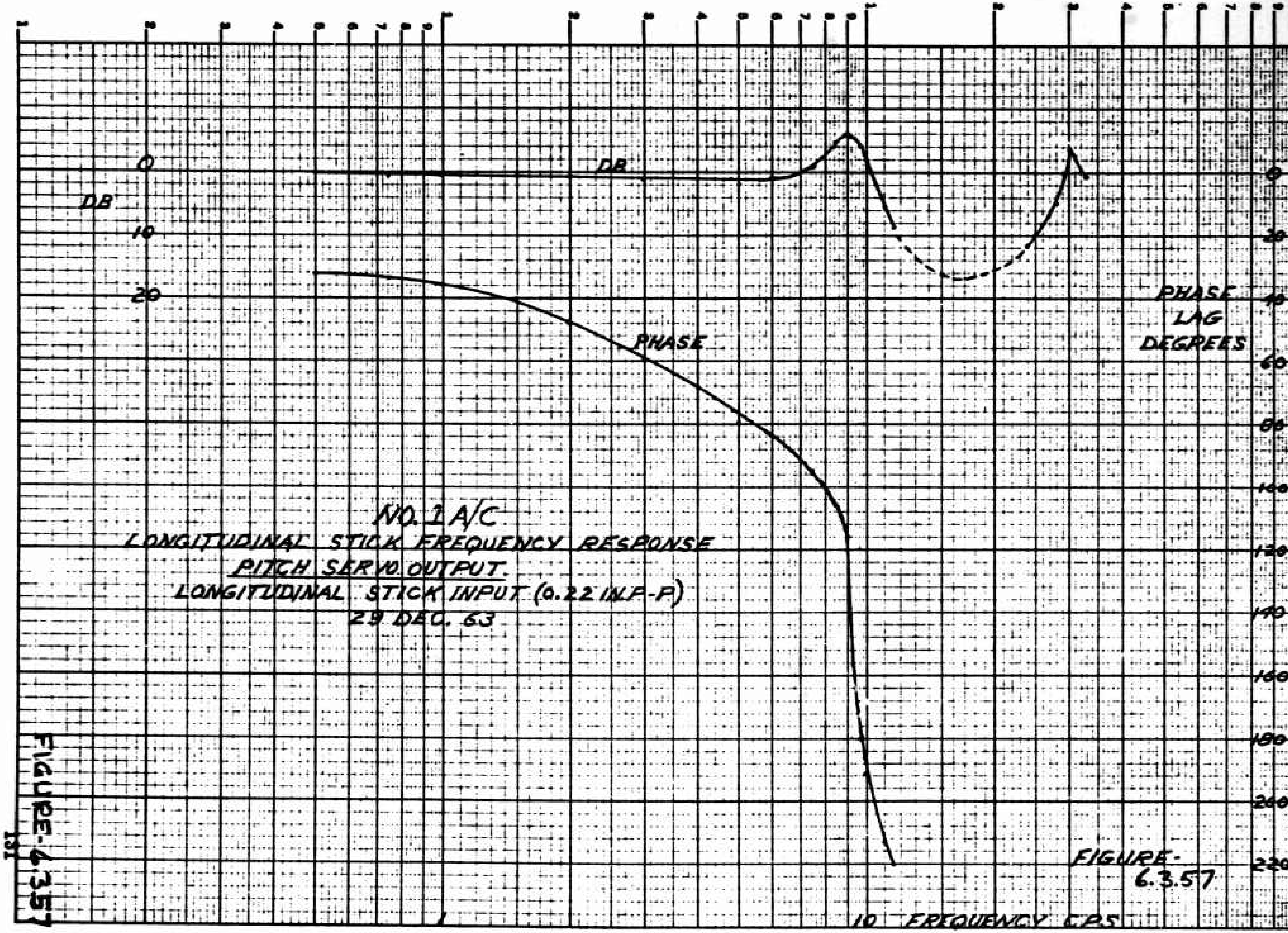

DB
PHASE
PHASE
LAG
DEGREES
NO. 1 A/C
LATERAL STICK FREQUENCY RESPONSE
COMPOSITE CURVE
L.H. AFT LOUVER SERVO OUTPUT
LATERAL STICK INPUT
21 JAN. 64
FIGURE-
6.3.55
1
10 FREQUENCY CPS
DB
10
20
20
40
60
80
100
120
140
160
180
200
220

FIGURE- 6.3.55



DB
PHASE
PHASE LAG DEGREES
NO. 1 A/C
LATERAL STICK FREQUENCY RESPONSE
COMPOSITE CURVE
L.H. FWD LOUVER SERVO OUTPUT
LATERAL STICK INPUT
21 JAN. 64
FIGURE-6.3.56
10 FREQUENCY CPS

FIGURE-6.3.56



NO. 1 A/C
LONGITUDINAL STICK FREQUENCY RESPONSE
PITCH SERVO OUTPUT
LONGITUDINAL STICK INPUT (0.22 IN.F-P)
29 DEC. 63
DB
PHASE
PHASE LAG DEGREES
10 FREQUENCY CPS
FIGURE-6.3.57

FIGURE-6.357

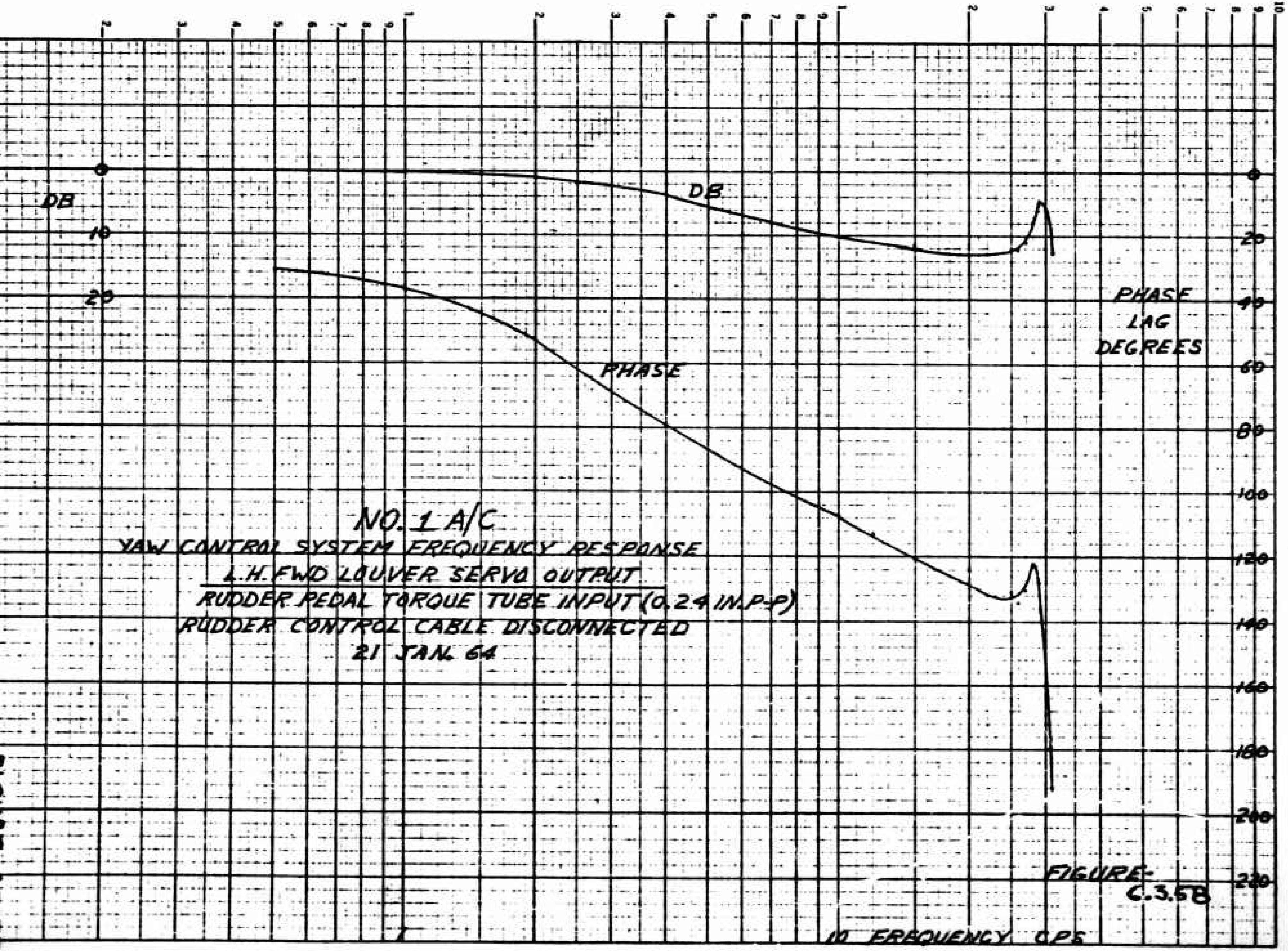

NO. 1 A/C
YAW CONTROL SYSTEM FREQUENCY RESPONSE
L.H. FWD LOUVER SERVO OUTPUT
RUDDER PEDAL TORQUE TUBE INPUT (0.24 IN. P-P)
RUDDER CONTROL CABLE DISCONNECTED
21 JAN. 64
DB
PHASE
PHASE LAG DEGREES
FIGURE- 6.3.58
FREQUENCY CPS

FIGURE- 6.3.58

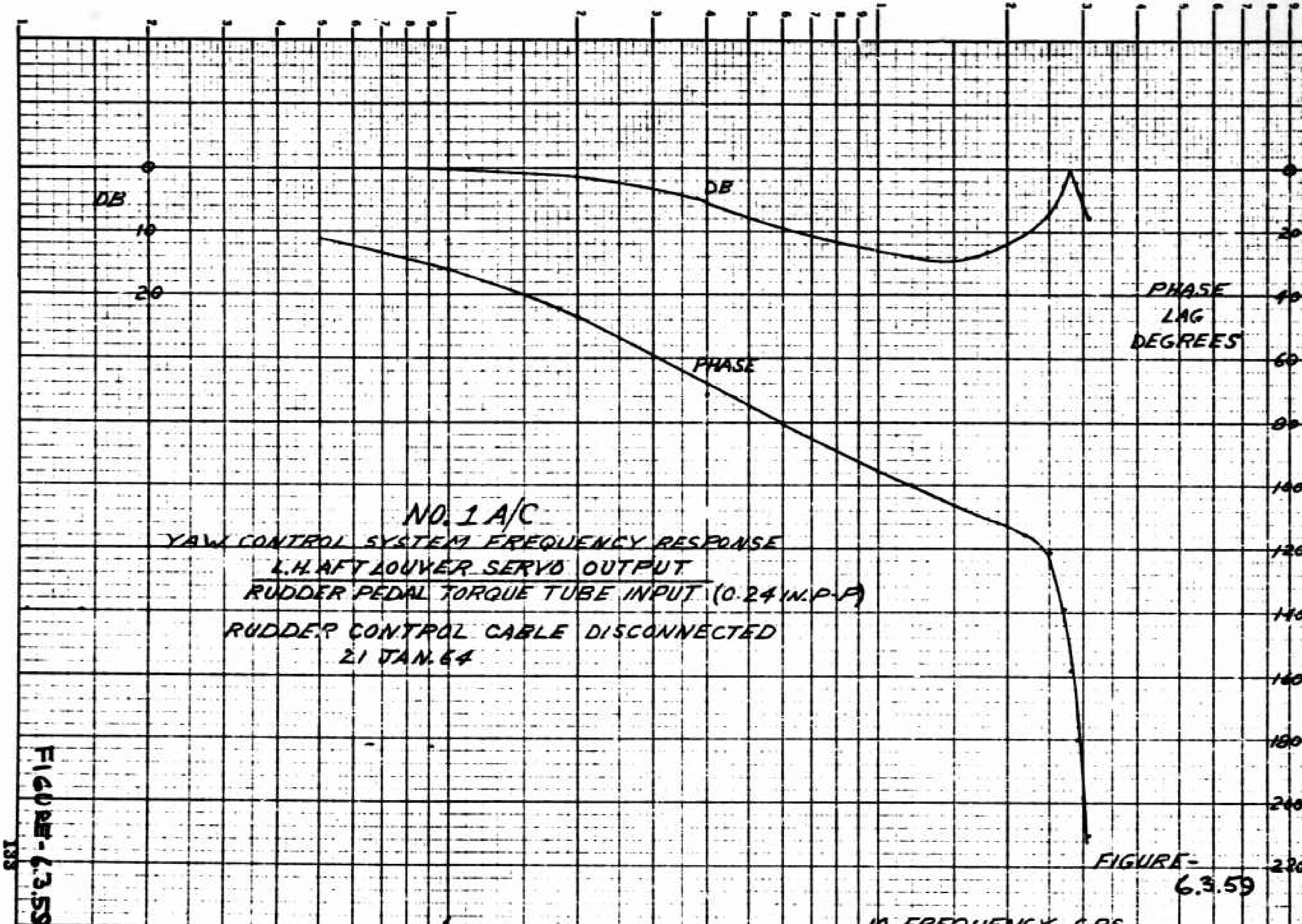
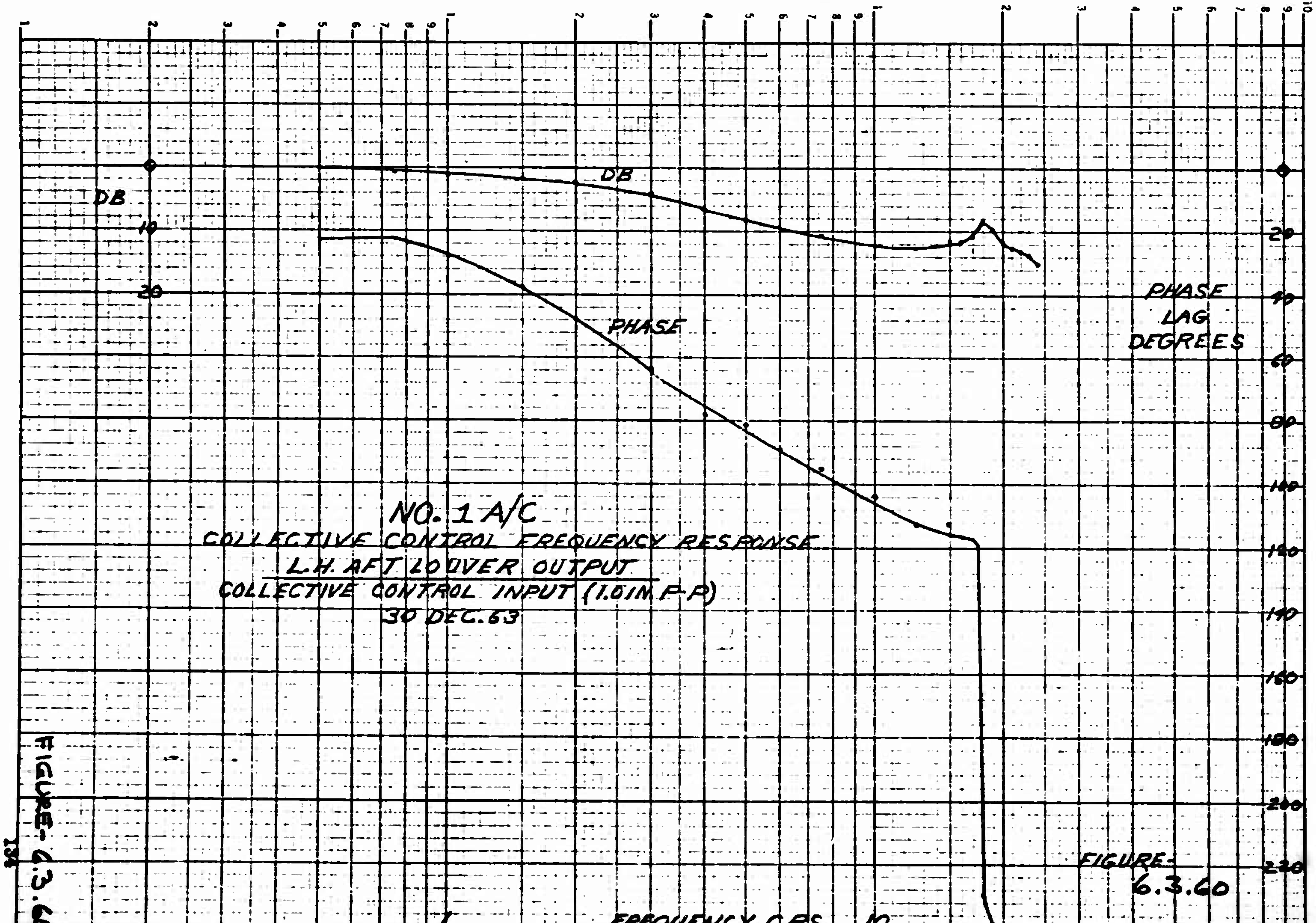

NO. 1 A/C
YAW CONTROL SYSTEM FREQUENCY RESPONSE
L.H. AFT LOUVER SERVO OUTPUT
RUDDER PEDAL TORQUE TUBE INPUT (0.24 IN. P-P)
RUDDER CONTROL CABLE DISCONNECTED
21 JAN. 64
DB
PHASE
PHASE LAG DEGREES
10 FREQUENCY C.P.S
FIGURE-6.3.59

FIGURE-6.3.59



NO. 1 A/C
COLLECTIVE CONTROL FREQUENCY RESPONSE
L.H. AFT LOUVER OUTPUT
COLLECTIVE CONTROL INPUT (1.0 IN. P-P)
30 DEC. 63
DB
PHASE
PHASE LAG DEGREES
FREQUENCY C.P.S.
FIGURE-6.3.60

FIGURE-6.3.60

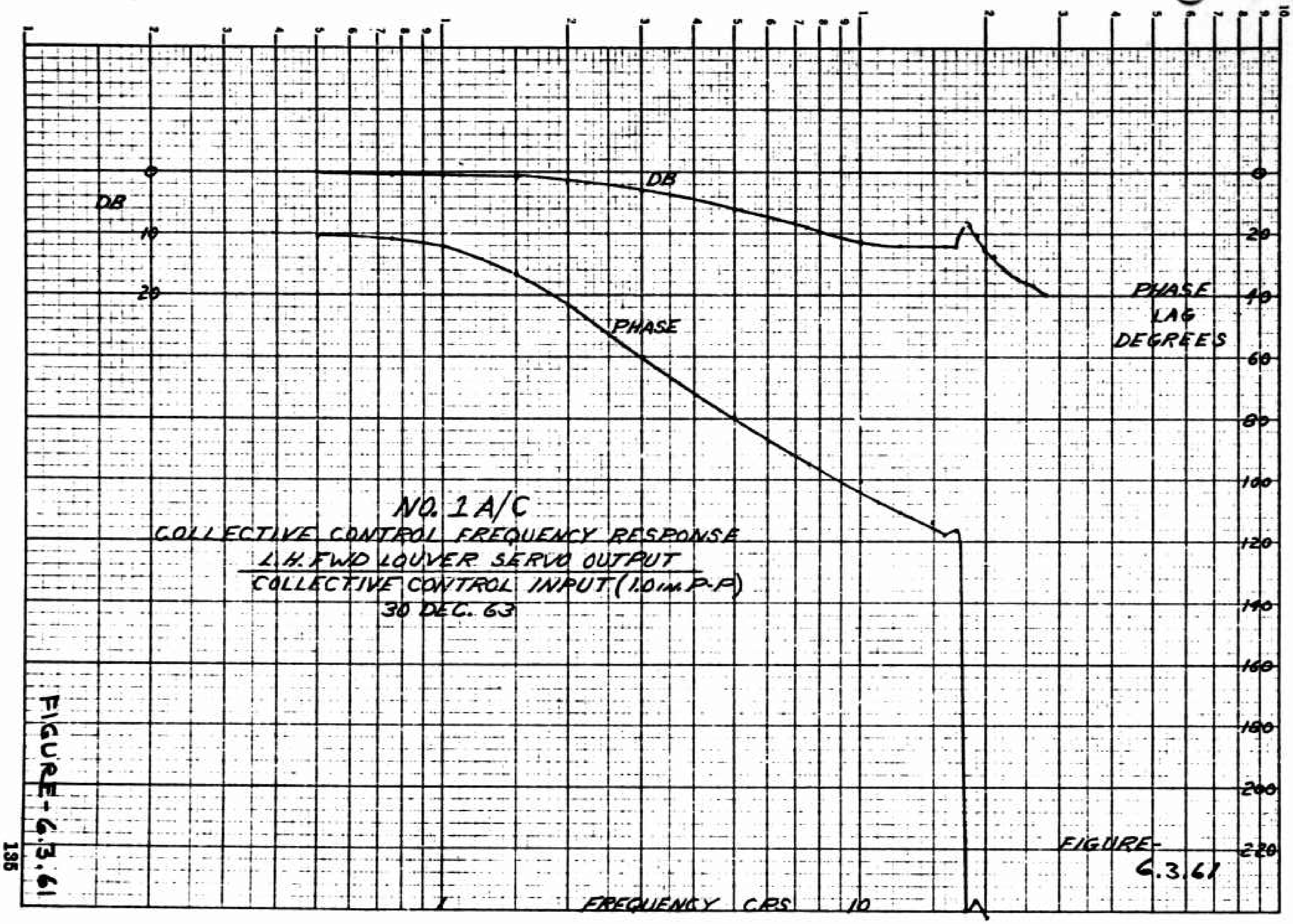

NO. 1 A/C
COLLECTIVE CONTROL FREQUENCY RESPONSE
L.H. FWD LOUVER SERVO OUTPUT
COLLECTIVE CONTROL INPUT (1.0 in. P-P)
30 DEC. 63
DB
PHASE
PHASE LAG DEGREES
0
20
40
60
80
100
120
140
160
180
200
220
DB
0
10
20
FREQUENCY CPS
1
10
FIGURE-6.3.61

FIGURE-6.3.61

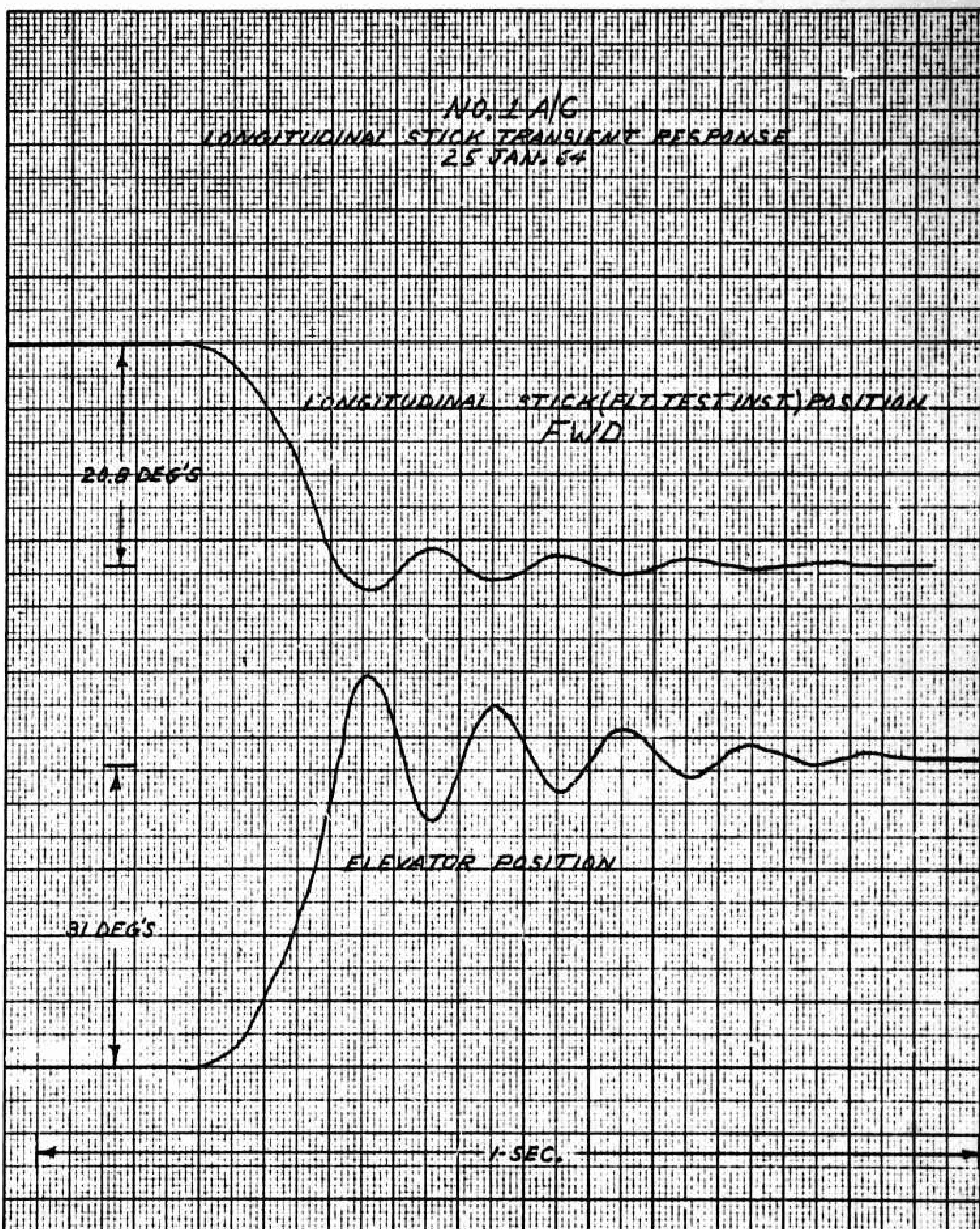

NO. 1 A/C
LONGITUDINAL STICK TRANSIENT RESPONSE
25 JAN. 64
LONGITUDINAL STICK (FLT TEST INST) POSITION
FWD
20.8 DEG'S
ELEVATOR POSITION
31 DEG'S
1 SEC.

FIGURE-6.3.62

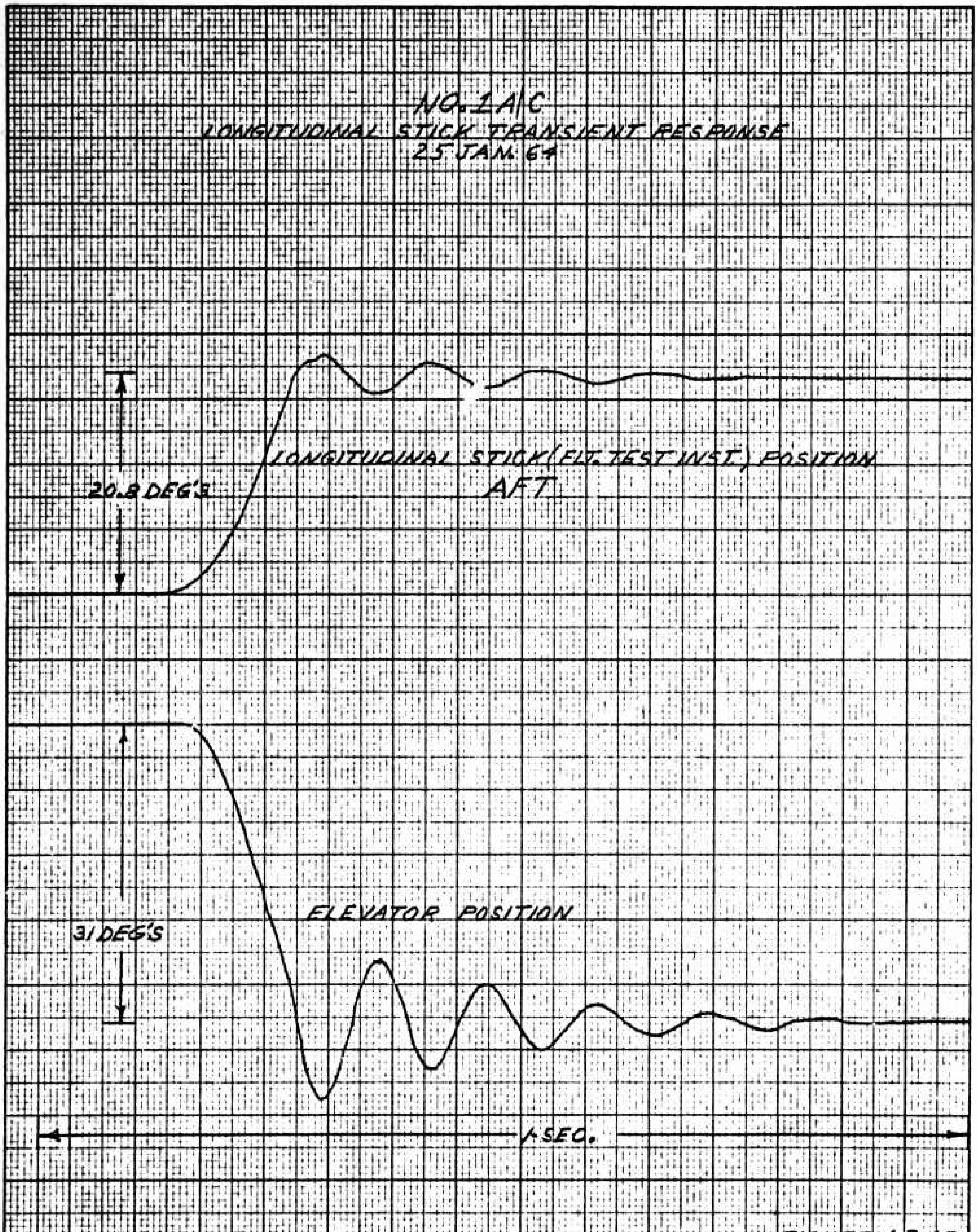
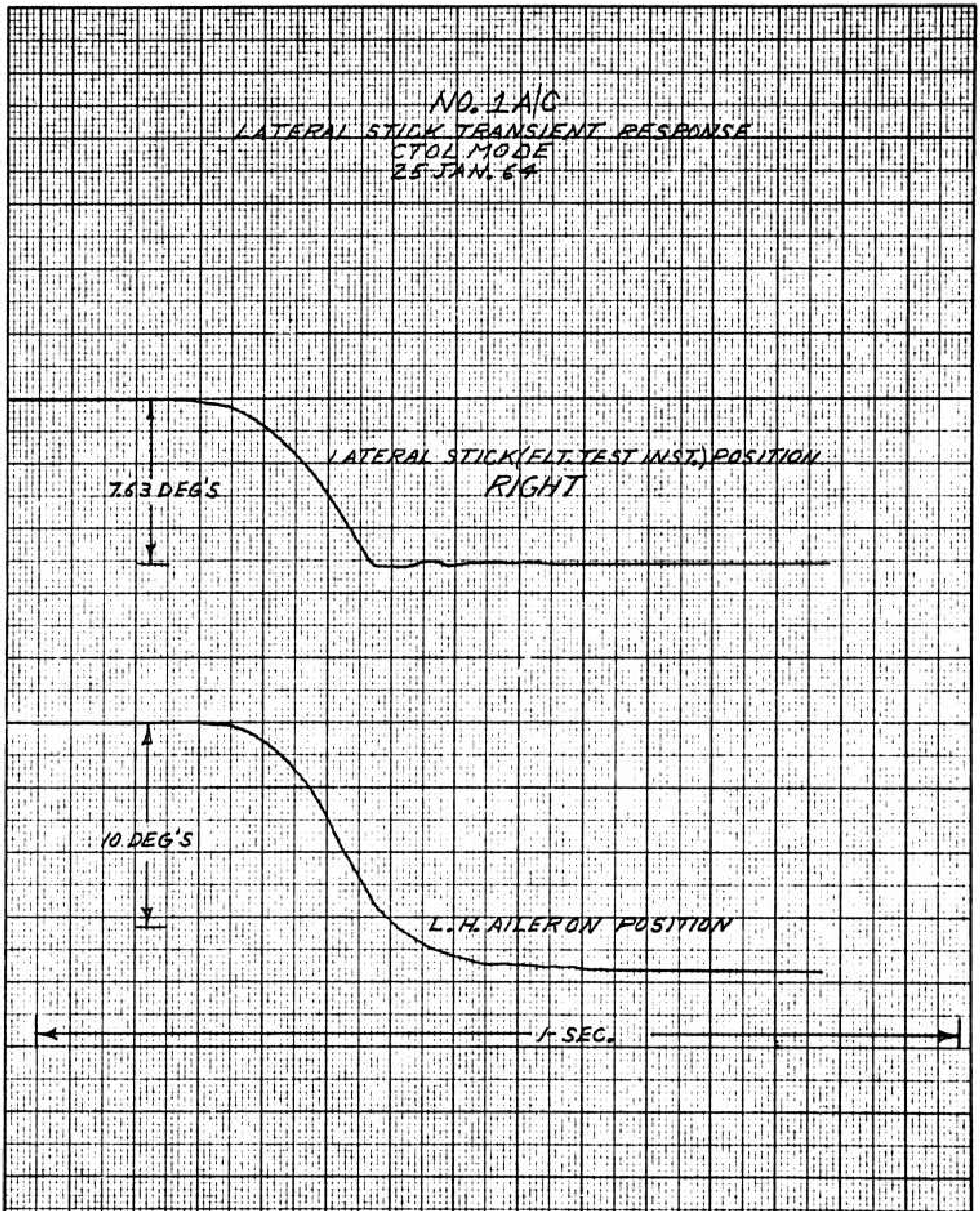

NO. 1 A/C
LONGITUDINAL STICK TRANSIENT RESPONSE
25 JAN. 64
20.8 DEG'S
LONGITUDINAL STICK (FLT. TEST INST.) POSITION
AFT
31 DEG'S
ELEVATOR POSITION
1 SEC.

FIGURE-63.63



NO. 1 A/C
LATERAL STICK TRANSIENT RESPONSE
CTOL MODE
25 JAN. 64
7.63 DEG'S
LATERAL STICK (FLT. TEST INST.) POSITION
RIGHT
10 DEG'S
L.H. AILERON POSITION
1 SEC.

FIGURE-6.3.64

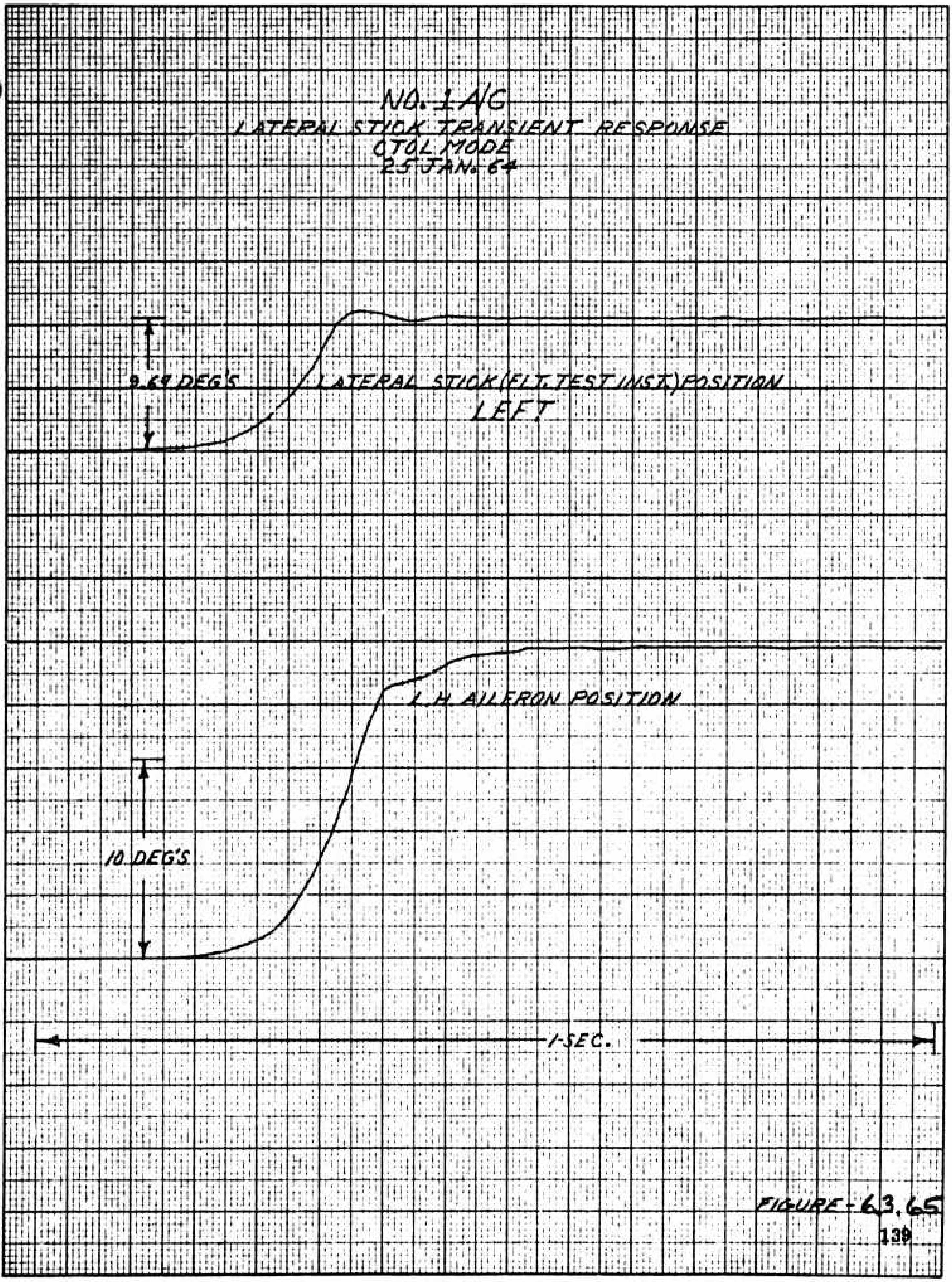

NO. 1 A/C
LATERAL STICK TRANSIENT RESPONSE
CTOL MODE
25 JAN. 64
9.69 DEG'S
LATERAL STICK (FLT. TEST INST.) POSITION
LEFT
L.H. AILERON POSITION
10 DEG'S
1-SEC.

FIGURE - 63.65

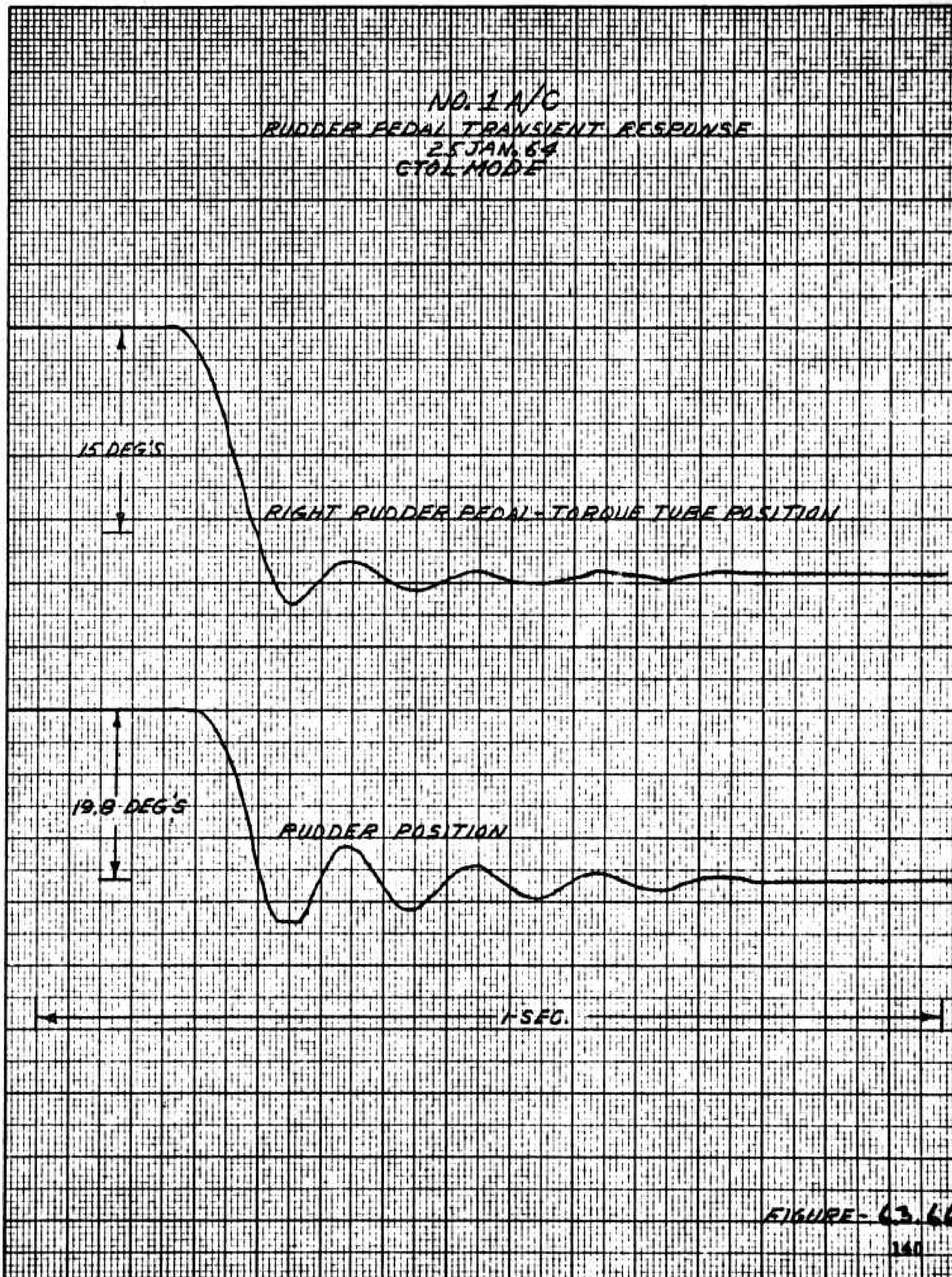
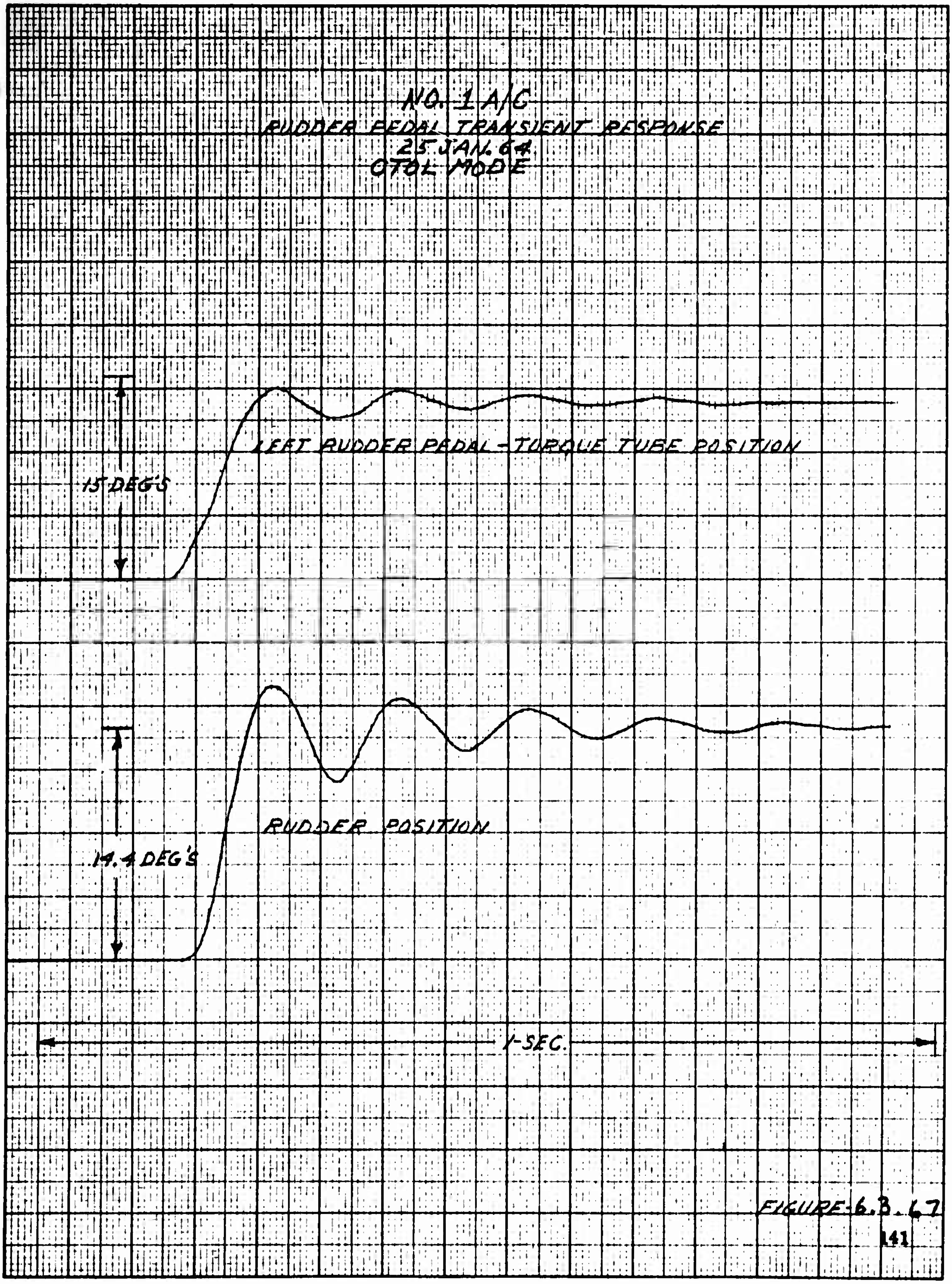

NO. 1 A/C
RUDDER PEDAL TRANSIENT RESPONSE
25 JAN. 64
CTOL MODE
15 DEG'S
RIGHT RUDDER PEDAL - TORQUE TUBE POSITION
19.8 DEG'S
RUDDER POSITION
1 SEC.

FIGURE- 63.66



NO. 1 A/C
RUDDER PEDAL TRANSIENT RESPONSE
25 JAN. 64
CTOL MODE
15 DEG'S
LEFT RUDDER PEDAL - TORQUE TUBE POSITION
14.4 DEG'S
RUDDER POSITION
1-SEC.

FIGURE 6.3.67

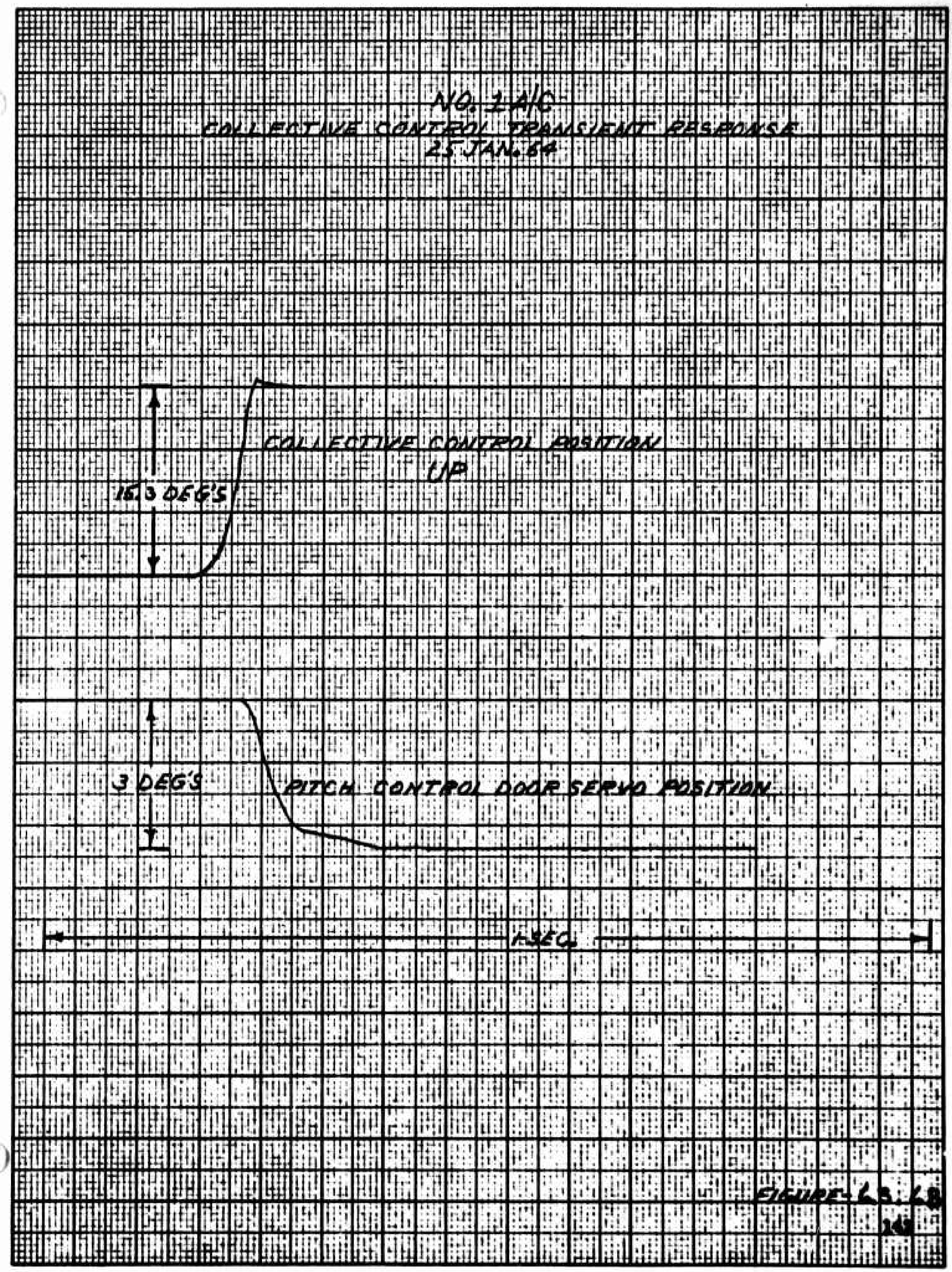

NO. 1 A/C
COLLECTIVE CONTROL TRANSIENT RESPONSE
25 JAN. 64
15.3 DEG'S
COLLECTIVE CONTROL POSITION
UP
3 DEG'S
PITCH CONTROL DOOR SERVO POSITION
1 SEC.

FIGURE- L3. L8

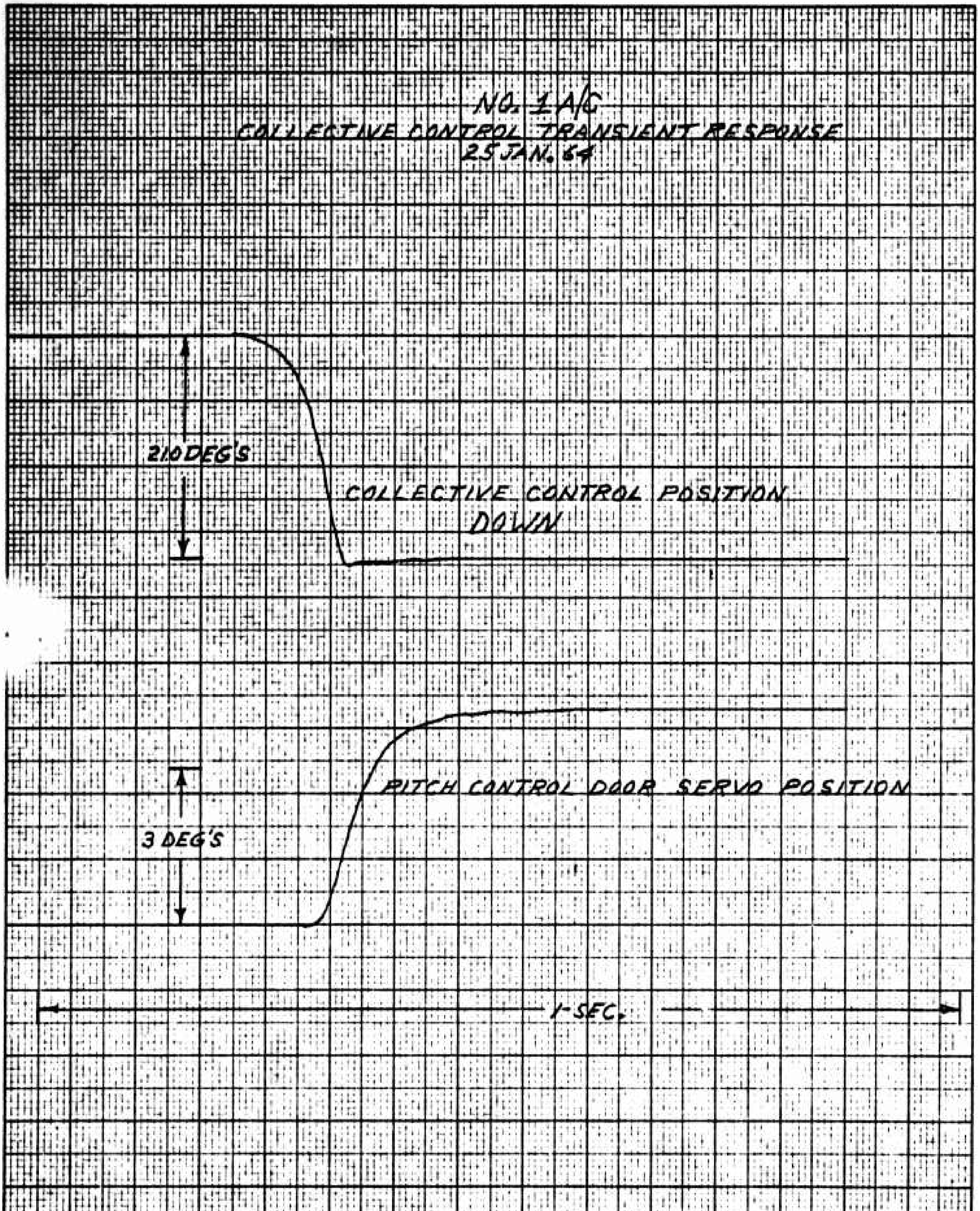

NO. 1 A/C
COLLECTIVE CONTROL TRANSIENT RESPONSE
25 JAN. 64
21.0 DEG'S
COLLECTIVE CONTROL POSITION
DOWN
3 DEG'S
PITCH CONTROL DOOR SERVO POSITION
1-SEC.

FIGURE-6.3.69

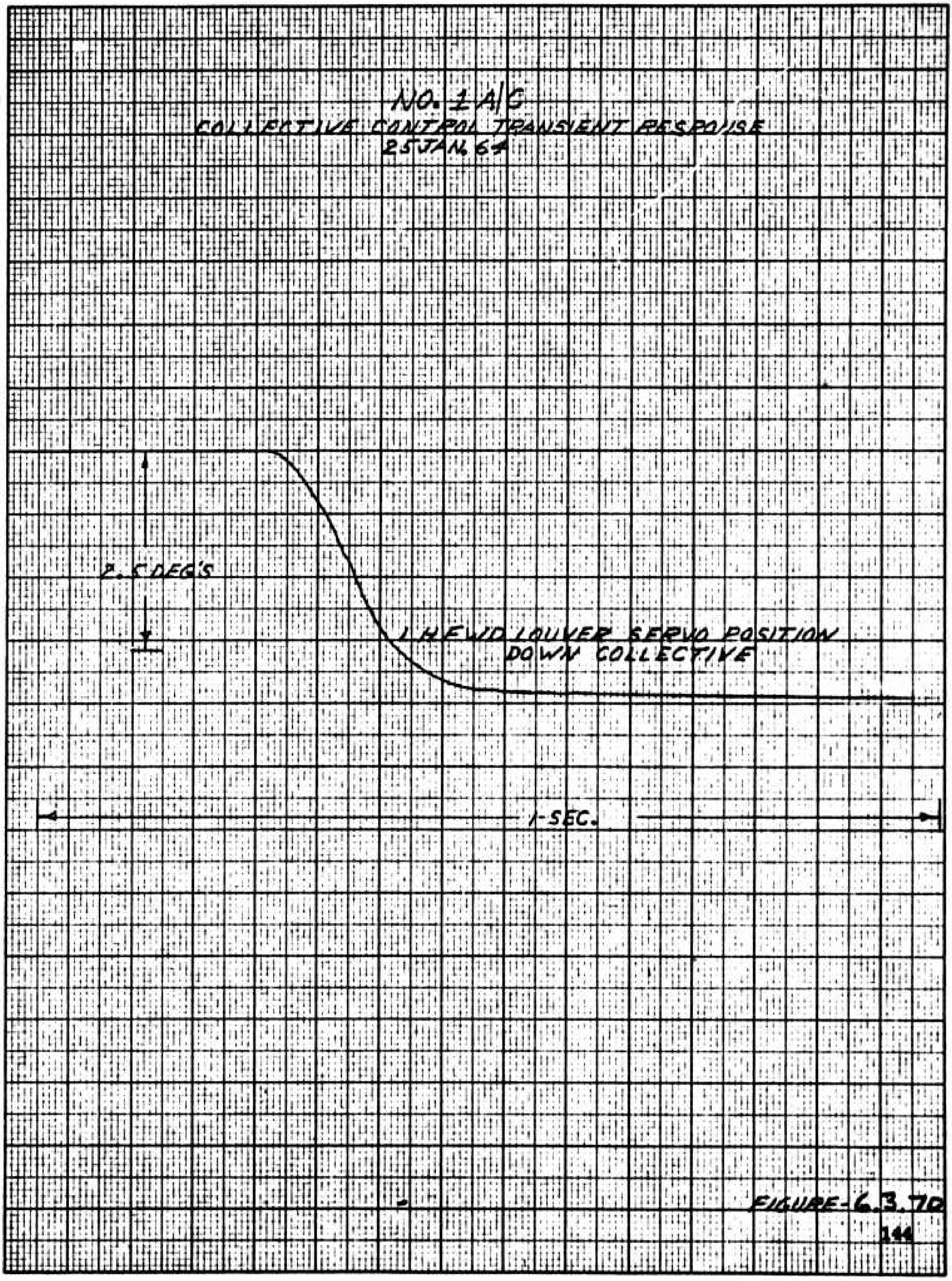

NO. 1 A/C
COLLECTIVE CONTROL TRANSIENT RESPONSE
25 JAN. 64
2.5 DEG'S
LH FWD LOUVER SERVO POSITION
DOWN COLLECTIVE
1-SEC.

FIGURE-6.3.70

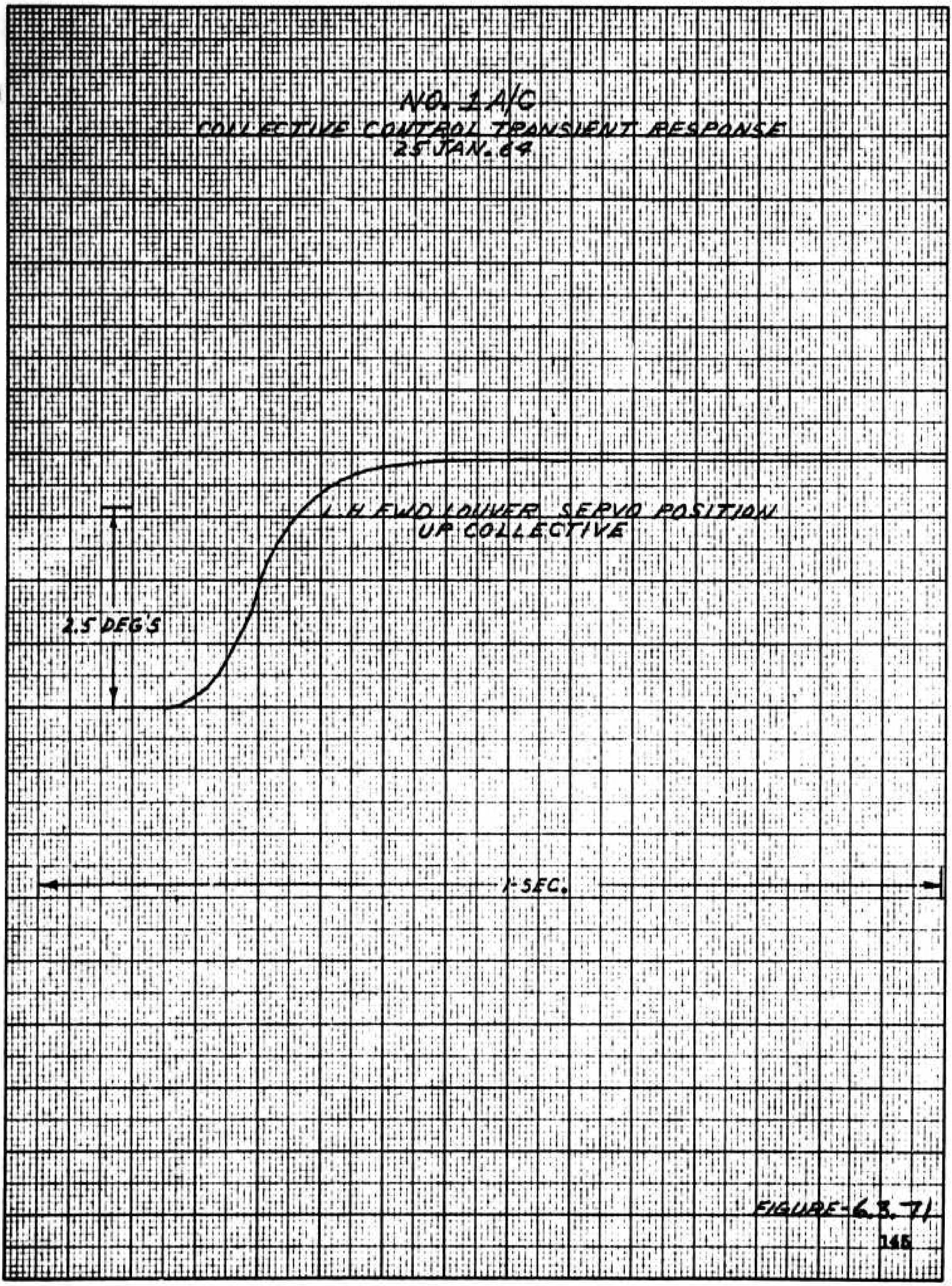

NO. 1 A/C
COLLECTIVE CONTROL TRANSIENT RESPONSE
25 JAN. 64
L.H FWD LOUVER SERVO POSITION
UP COLLECTIVE
2.5 DEG'S
1-SEC.

FIGURE-6.3.71

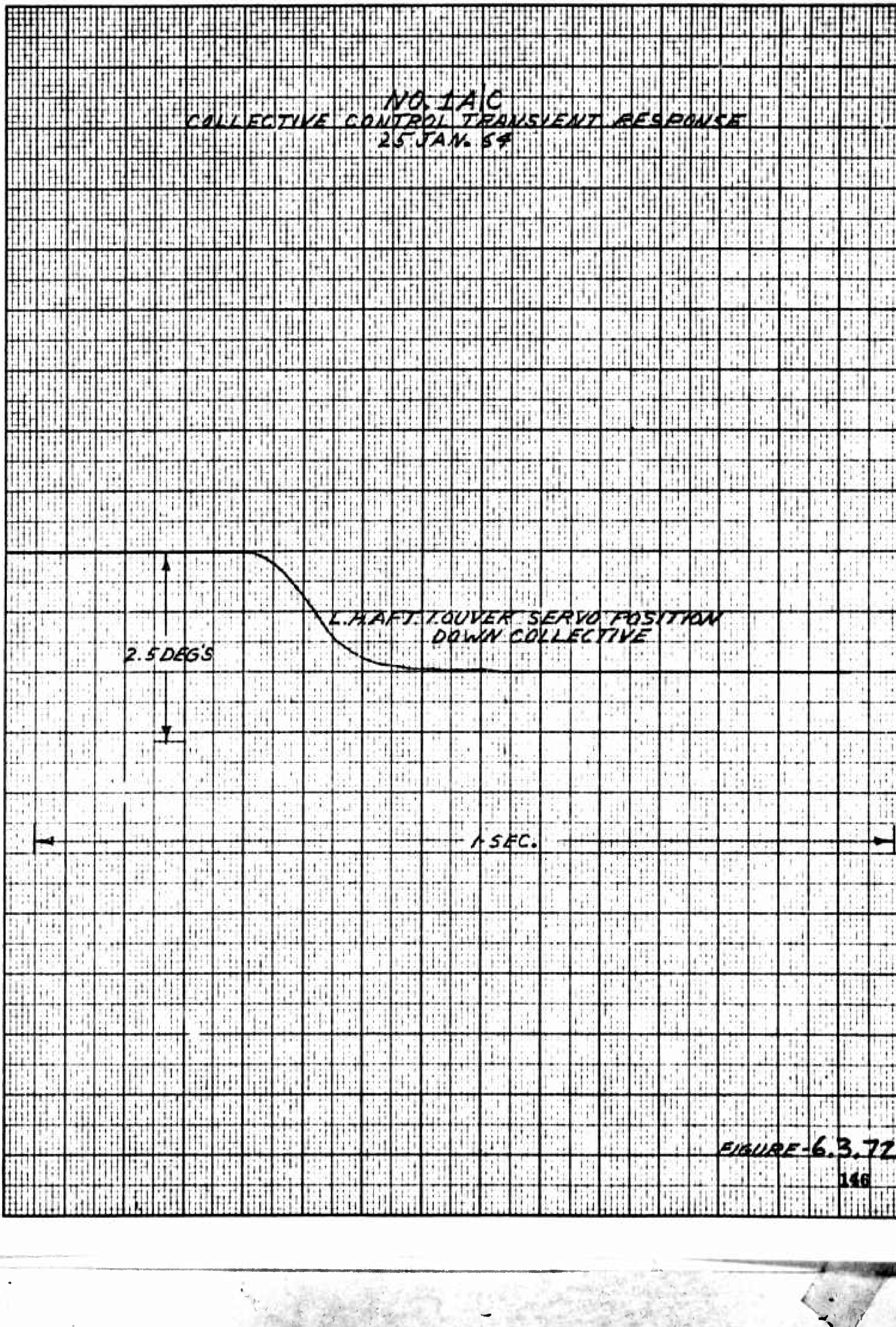

NO. 1A|C
COLLECTIVE CONTROL TRANSIENT RESPONSE
25 JAN. 54
L.H. AFT. LOUVER SERVO POSITION
DOWN COLLECTIVE
2.5 DEG'S
1 SEC.

FIGURE-6.3.72

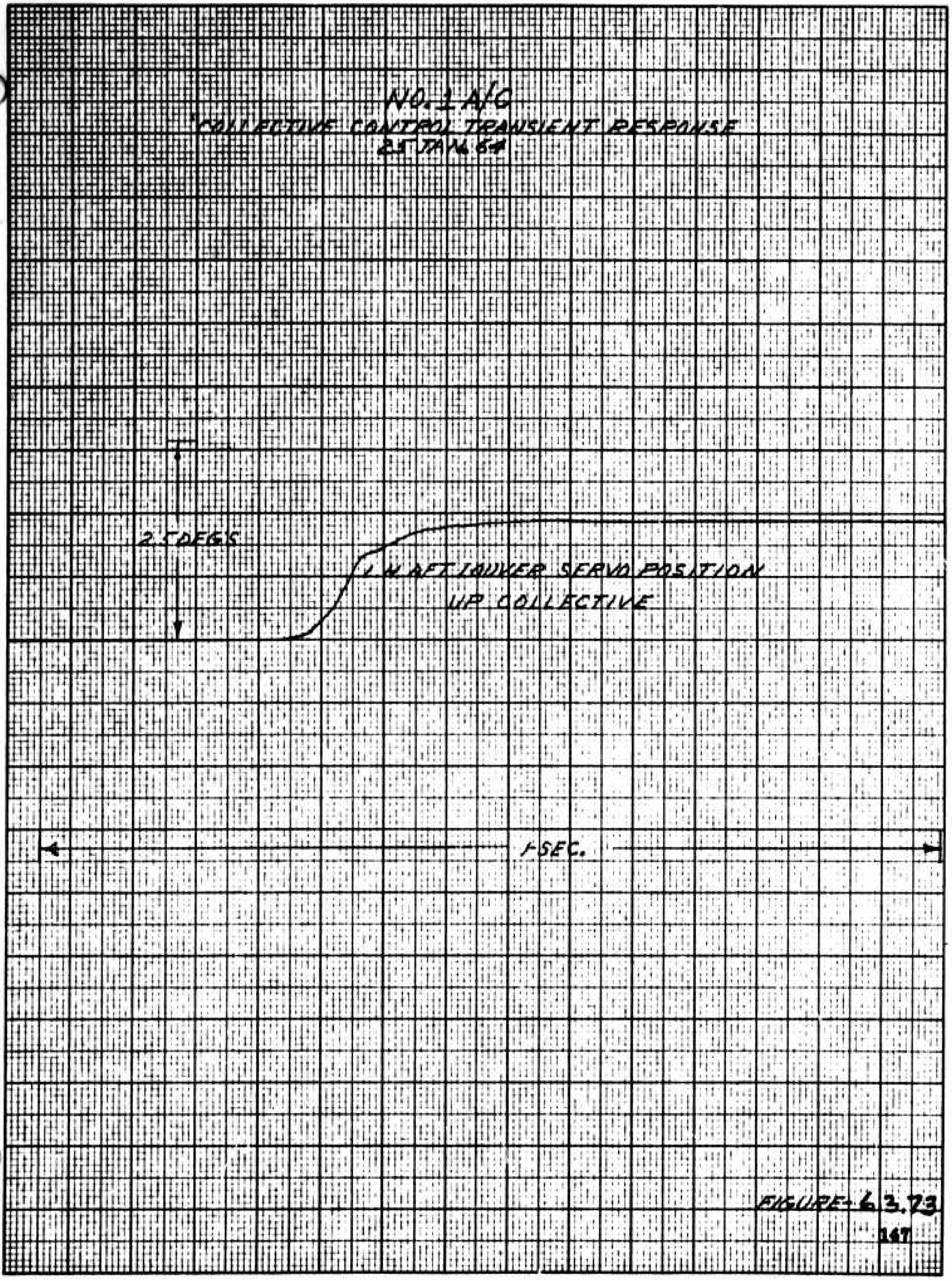

NO. 1 A/C
COLLECTIVE CONTROL TRANSIENT RESPONSE
25 JAN. 64
2.5 DEGS
L.H. AFT LOUVER SERVO POSITION
UP COLLECTIVE
1 SEC.

FIGURE- 6.3.73

NO. 1 A/C
LATERAL STICK TRANSIENT RESPONSE
VTOL MODE
25 JAN. 64

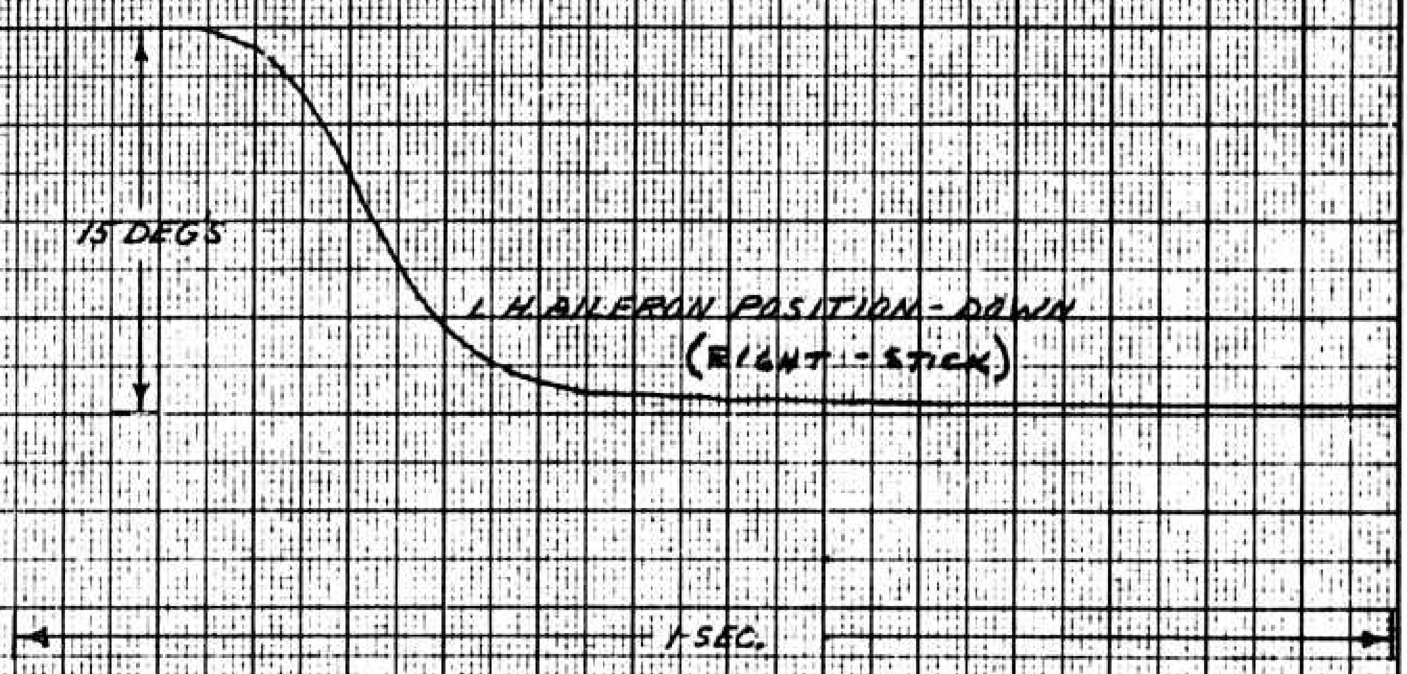


FIGURE 6.3.74

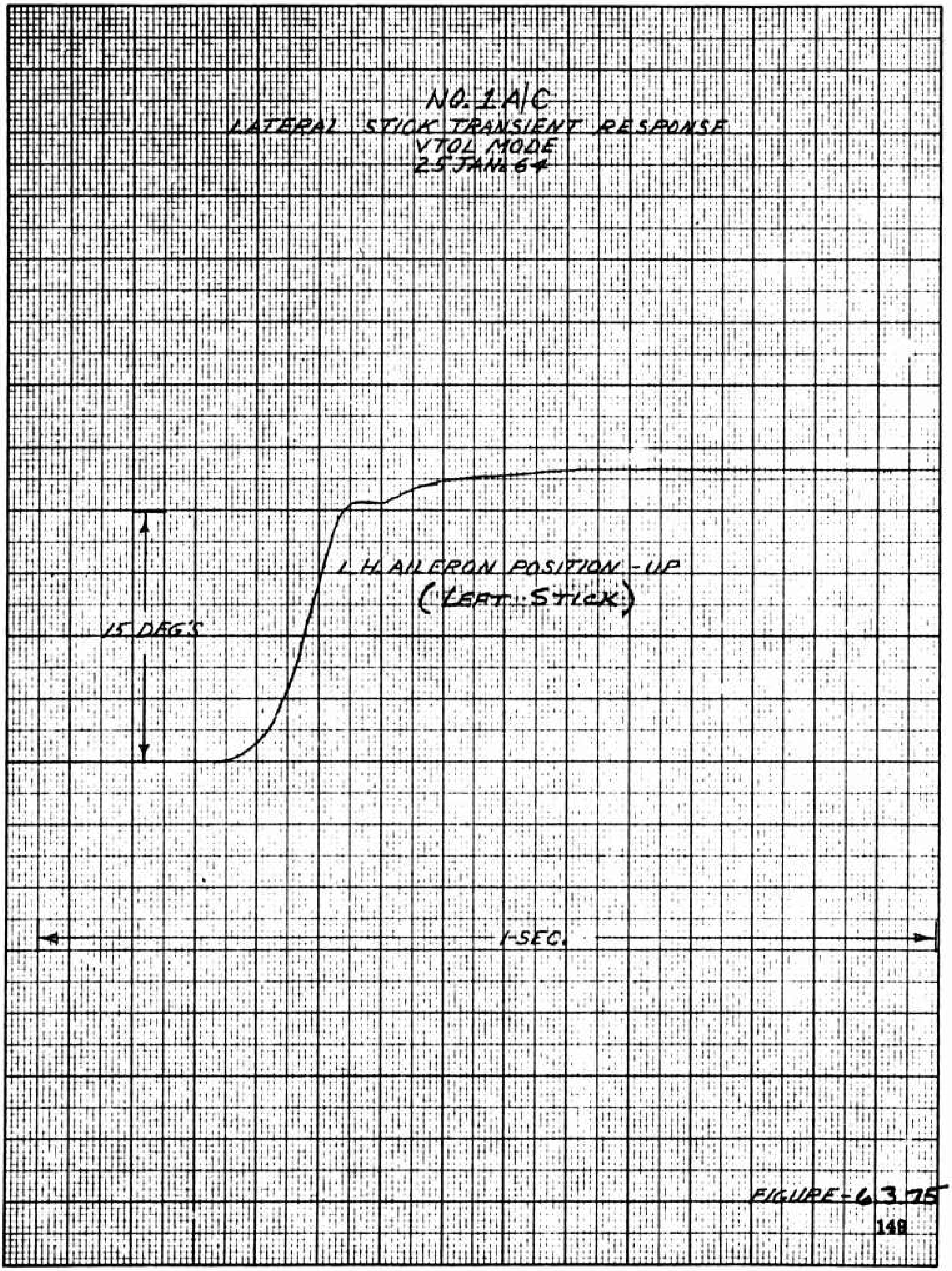

NO. 1 A/C
LATERAL STICK TRANSIENT RESPONSE
VTOL MODE
25 JAN. 64
L.H. AILERON POSITION - UP
(LEFT STICK)
15 DEG'S
1-SEC.

FIGURE-6.3.75

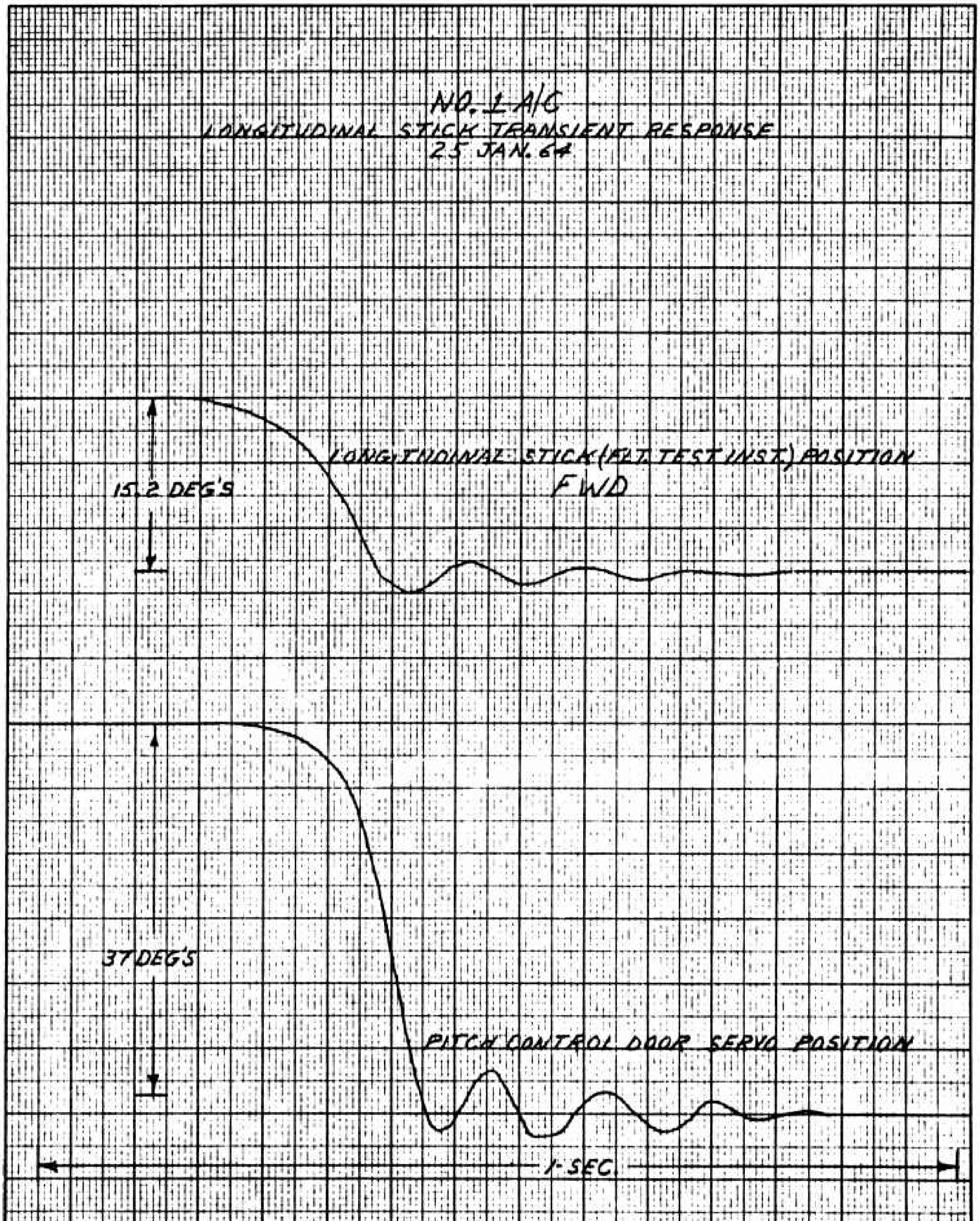

NO. 1 A/C
LONGITUDINAL STICK TRANSIENT RESPONSE
25 JAN. 64
LONGITUDINAL STICK (FLT. TEST INST.) POSITION
FWD
15.2 DEG'S
37 DEG'S
PITCH CONTROL DOOR SERVO POSITION
1-SEC.

FIGURE - 6.3.76

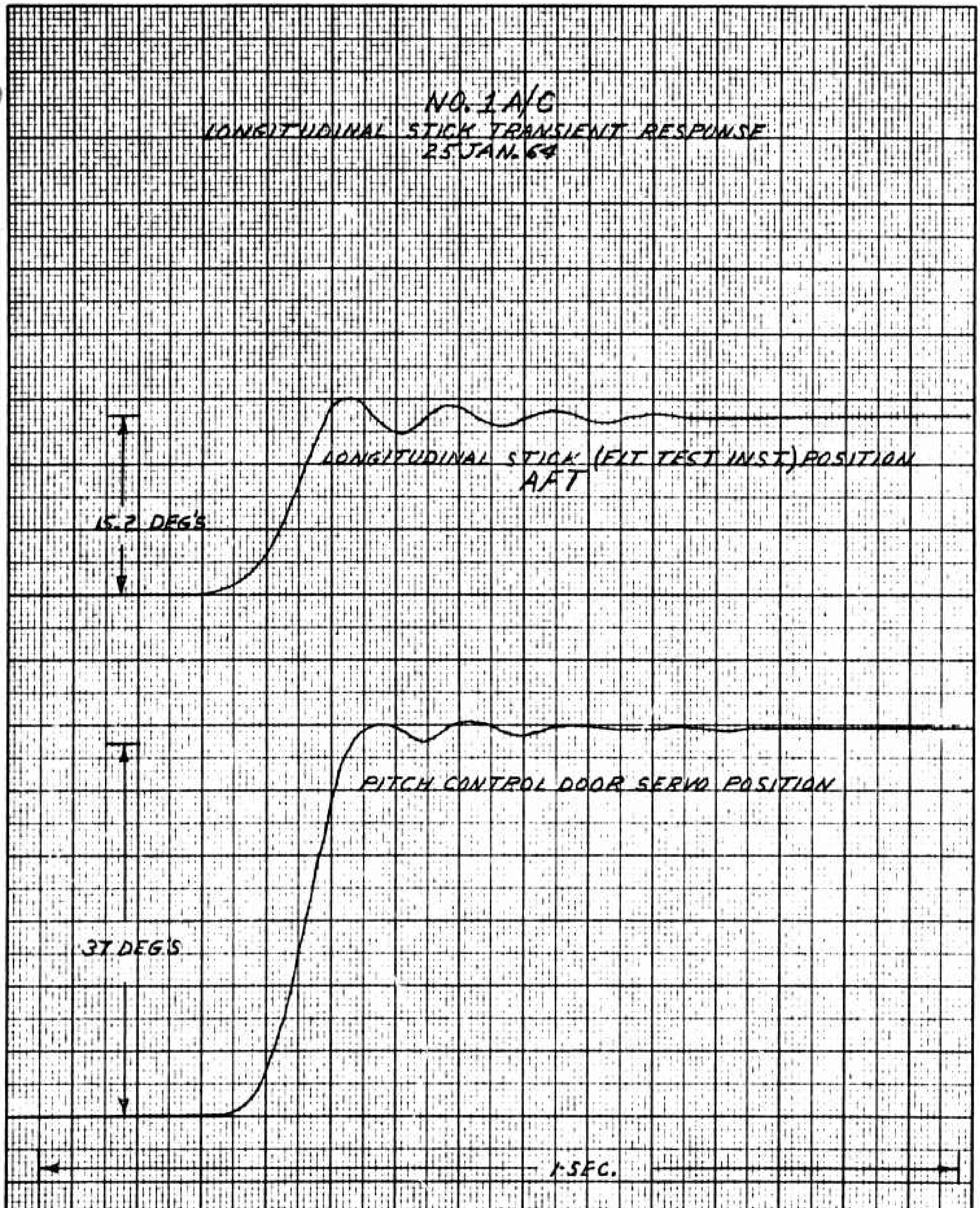

NO. 1 A/C
LONGITUDINAL STICK TRANSIENT RESPONSE
25 JAN. 64
LONGITUDINAL STICK (FLT TEST INST) POSITION
AFT
15.2 DEG'S
PITCH CONTROL DOOR SERVO POSITION
37 DEG'S
1 SEC.

FIGURE-6.3.77

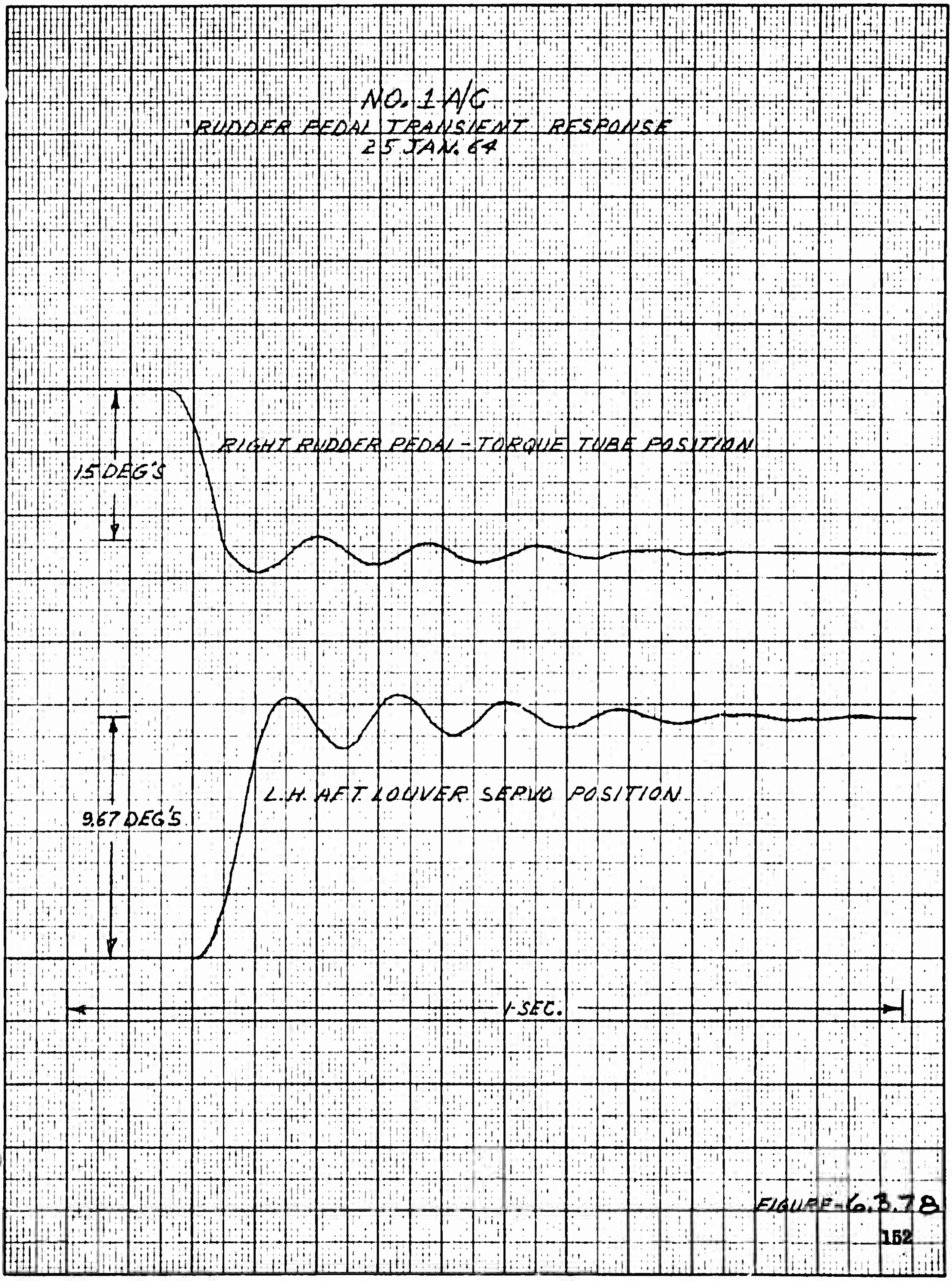

NO. 1 A/C
RUDDER PEDAL TRANSIENT RESPONSE
25 JAN. 64
RIGHT RUDDER PEDAL - TORQUE TUBE POSITION
15 DEG'S
L.H. AFT LOUVER SERVO POSITION
9.67 DEG'S
1-SEC.

FIGURE-6.3.78

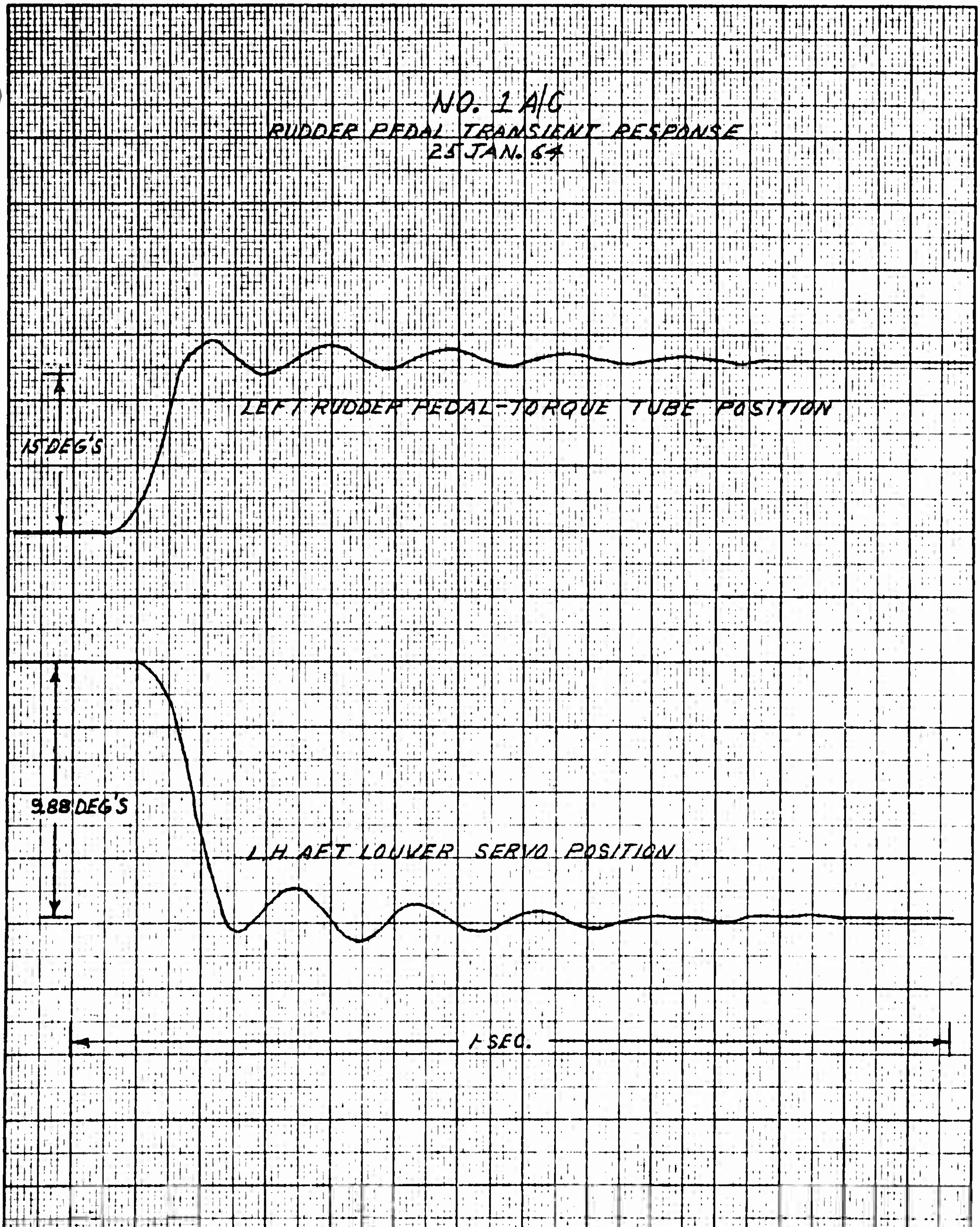

NO. 1 A/C
RUDDER PEDAL TRANSIENT RESPONSE
25 JAN. 64
15 DEG'S
LEFT RUDDER PEDAL-TORQUE TUBE POSITION
9.88 DEG'S
L.H. AFT LOUVER SERVO POSITION
1 SEC.

FIGURE-6.3.79

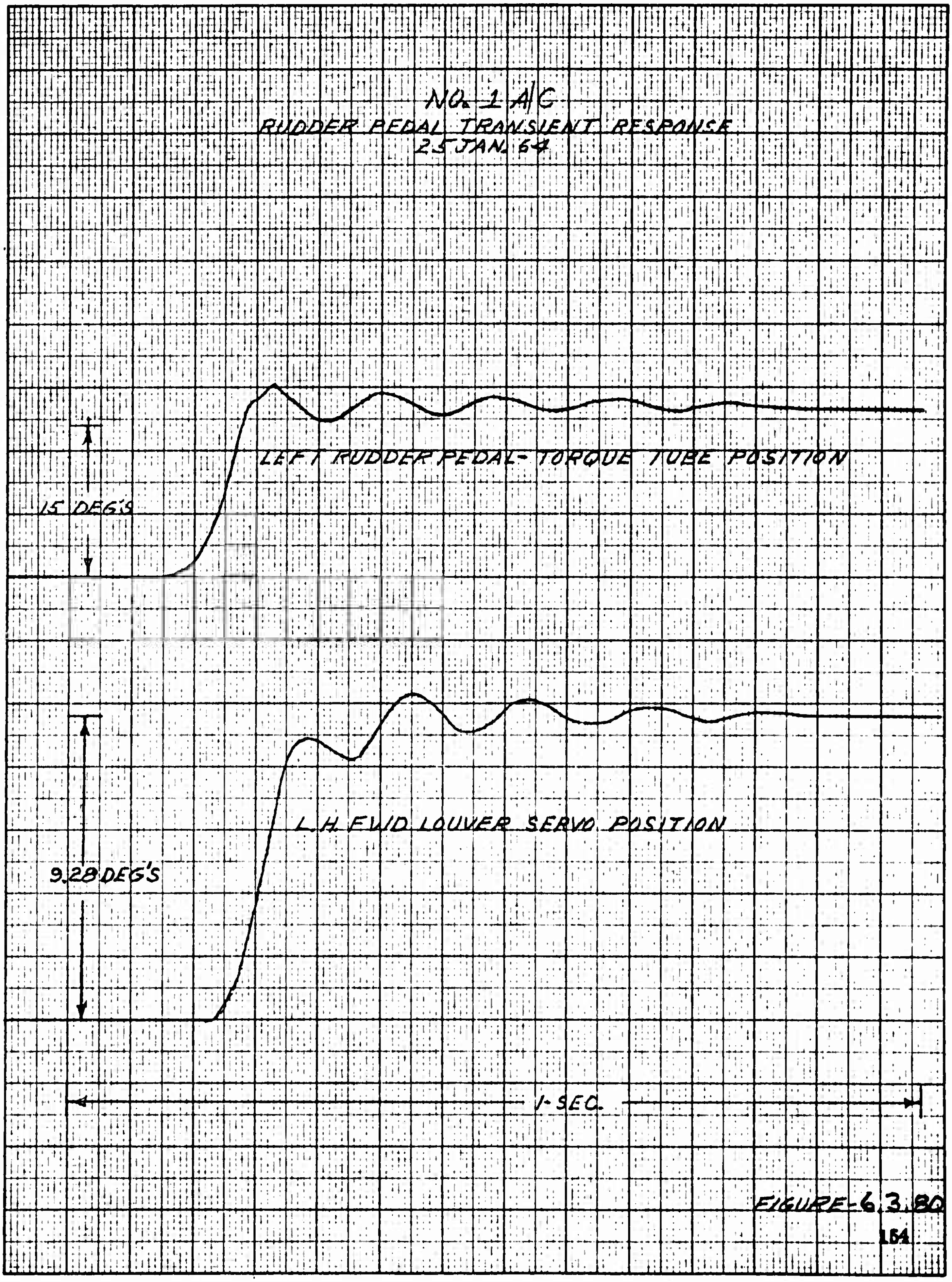

NO. 1 A/C
RUDDER PEDAL TRANSIENT RESPONSE
25 JAN. 64
LEFT RUDDER PEDAL- TORQUE TUBE POSITION
15 DEG'S
L.H. FWD LOUVER SERVO POSITION
9.28 DEG'S
1-SEC.

FIGURE-6.3.80

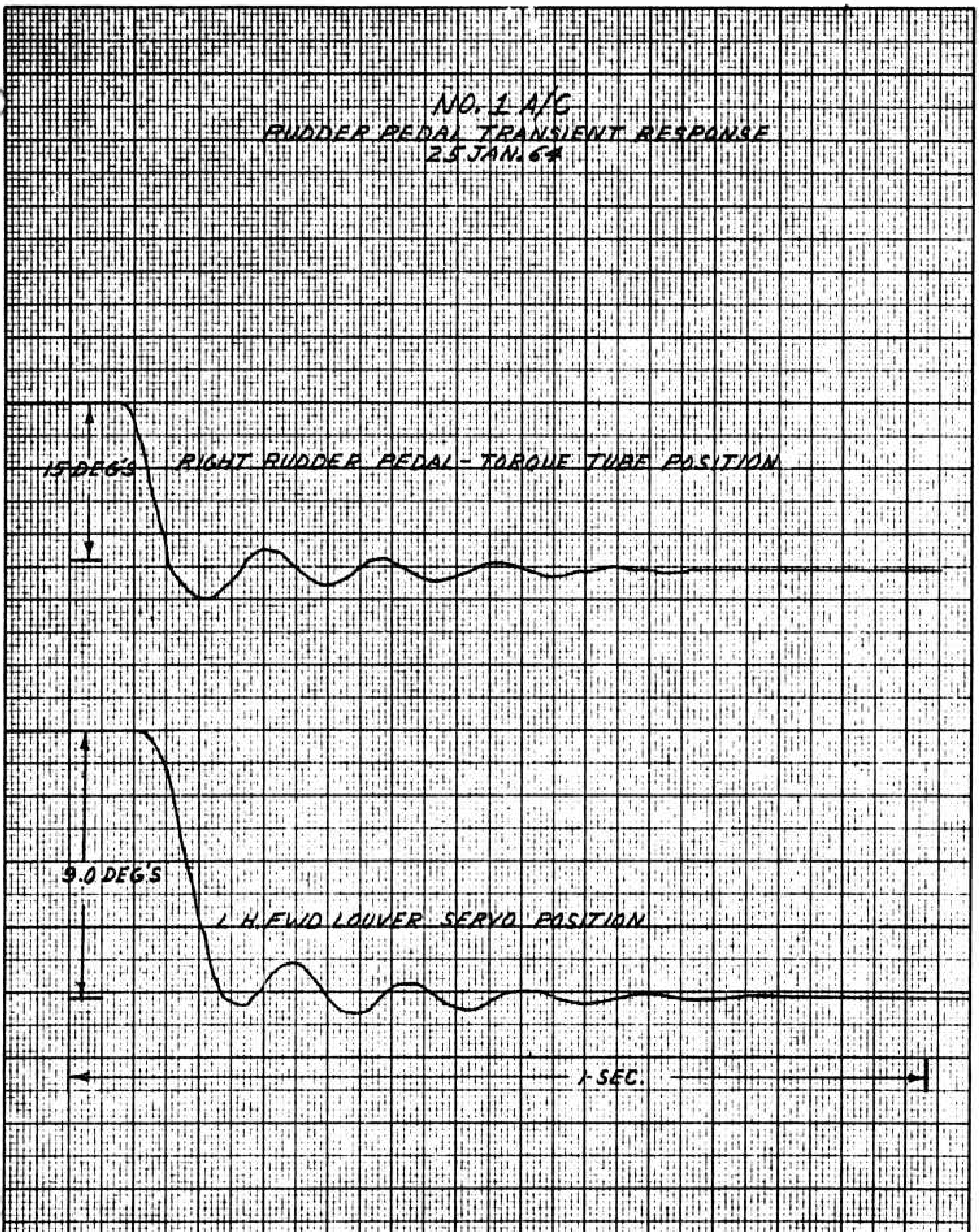
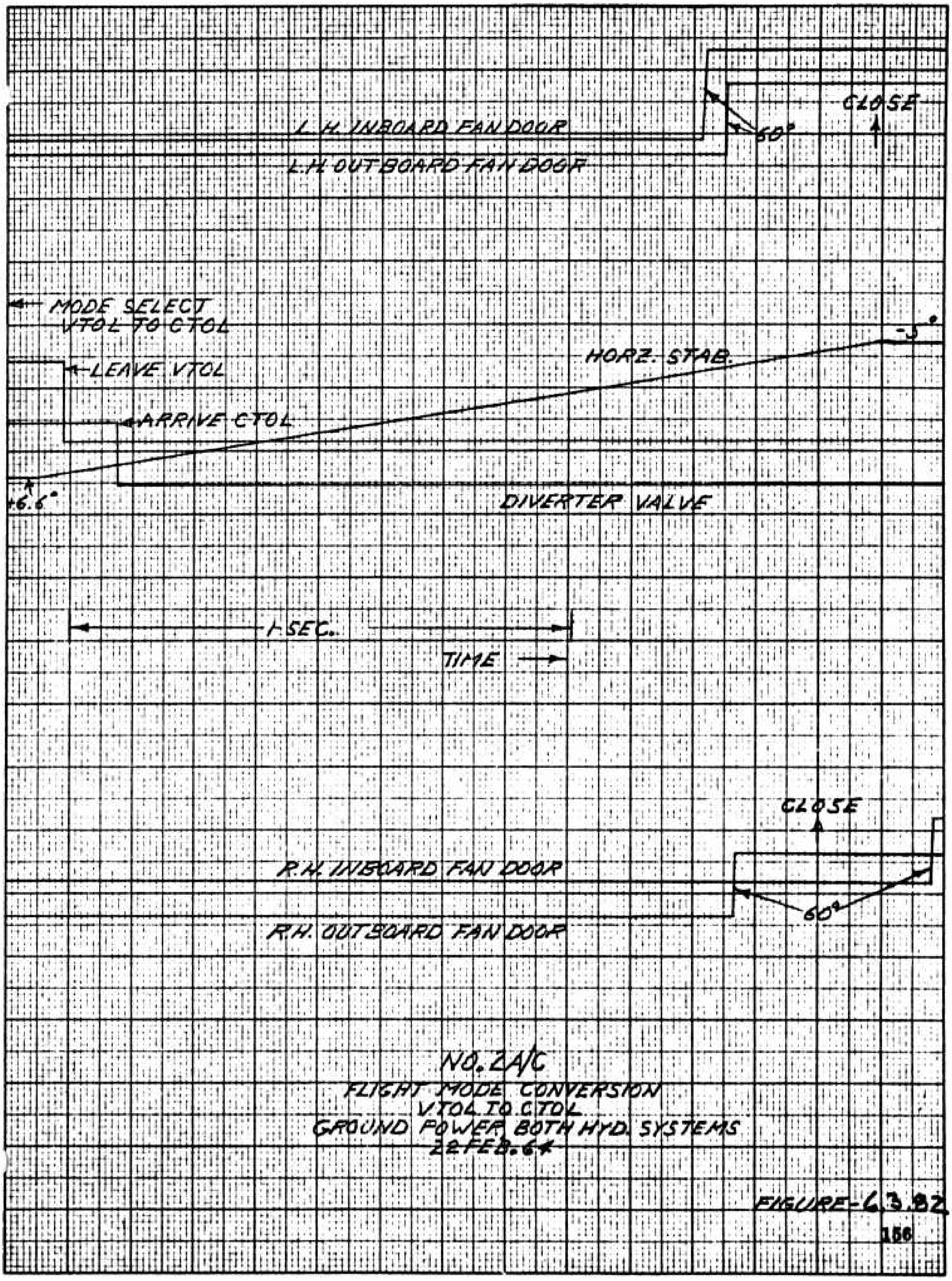

NO. 1 A/C
RUDDER PEDAL TRANSIENT RESPONSE
25 JAN. 64
15 DEG'S
RIGHT RUDDER PEDAL - TORQUE TUBE POSITION
9.0 DEG'S
L.H. FWD LOUVER SERVO POSITION
1-SEC.

FIGURE-6.3.9.1


CLOSE
L.H. INBOARD FAN DOOR
60°
L.H. OUTBOARD FAN DOOR
MODE SELECT
VTOL TO CTOL
-5°
HORZ. STAB.
LEAVE VTOL
ARRIVE CTOL
+6.6°
DIVERTER VALVE
1 SEC.
TIME
CLOSE
R.H. INBOARD FAN DOOR
60°
R.H. OUTBOARD FAN DOOR
NO. 2 A/C
FLIGHT MODE CONVERSION
VTOL TO CTOL
GROUND POWER BOTH HYD. SYSTEMS
22 FEB. 64
FIGURE-6.3.82

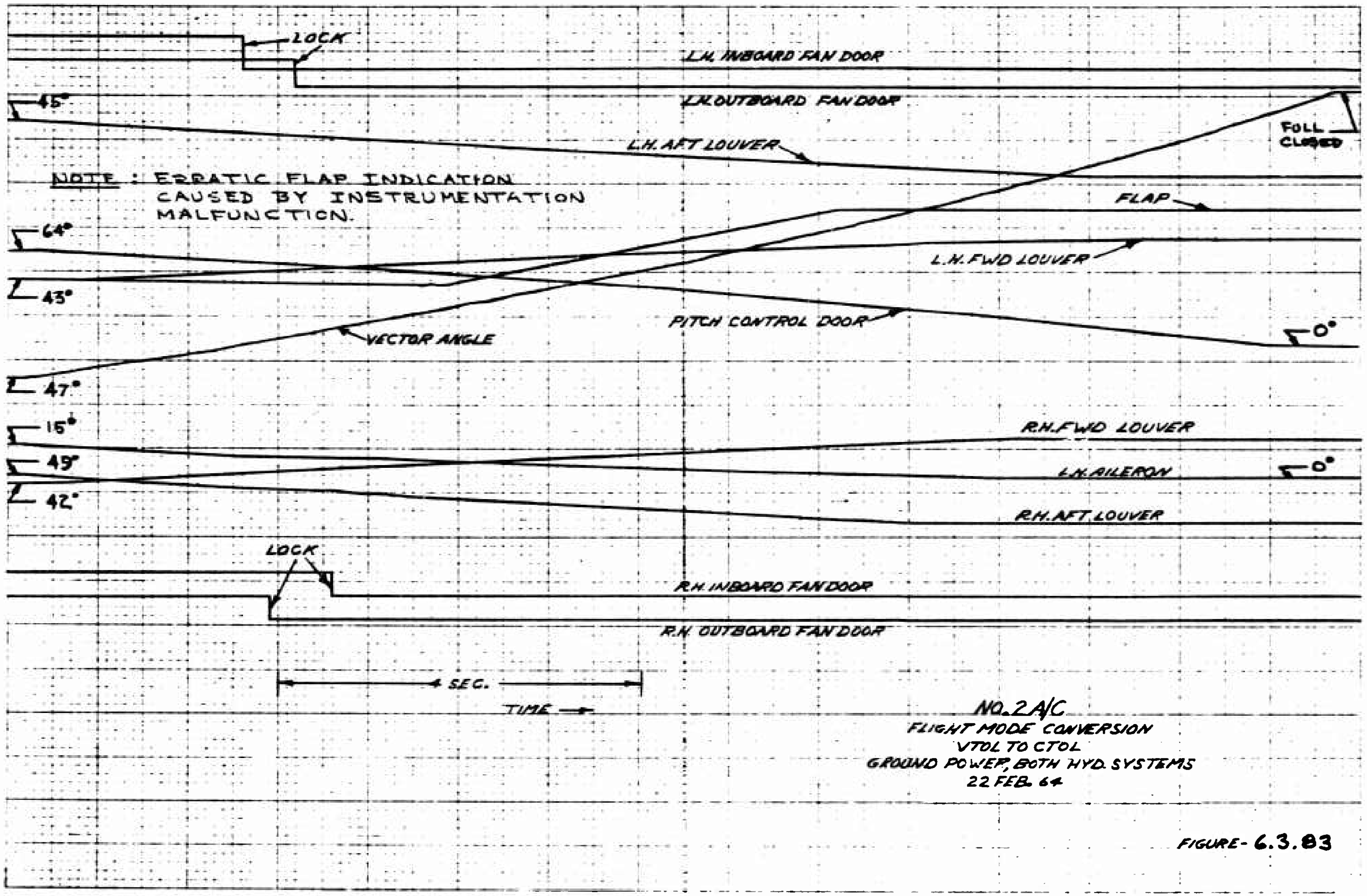

LOCK
L.H. INBOARD FAN DOOR
L.H. OUTBOARD FAN DOOR
45°
FULL CLOSED
L.H. AFT LOUVER
NOTE : ERRATIC FLAP INDICATION
CAUSED BY INSTRUMENTATION
MALFUNCTION.
FLAP
64°
L.H. FWD LOUVER
43°
PITCH CONTROL DOOR
VECTOR ANGLE
0°
47°
15°
R.H. FWD LOUVER
49°
L.H. AILERON
0°
42°
R.H. AFT LOUVER
LOCK
R.H. INBOARD FAN DOOR
R.H. OUTBOARD FAN DOOR
4 SEC.
TIME
NO. 2 A/C
FLIGHT MODE CONVERSION
VTOL TO CTOL
GROUND POWER, BOTH HYD. SYSTEMS
22 FEB. 64

FIGURE- 6.3.83

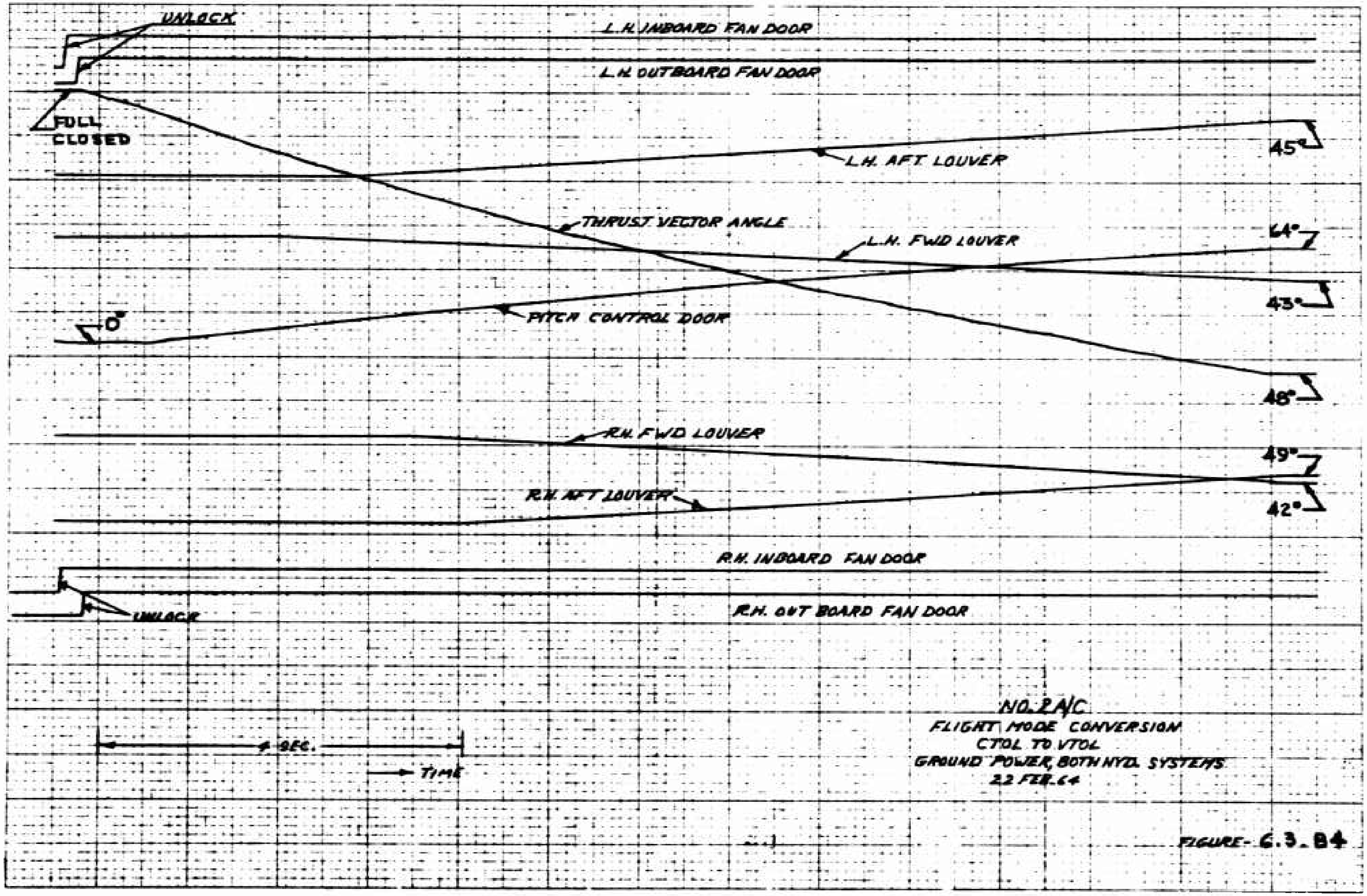

UNLOCK
L.H. INBOARD FAN DOOR
L.H. OUTBOARD FAN DOOR
FULL CLOSED
45°
L.H. AFT LOUVER
THRUST VECTOR ANGLE
64°
L.H. FWD LOUVER
43°
0°
PITCH CONTROL DOOR
48°
R.H. FWD LOUVER
49°
R.H. AFT LOUVER
42°
R.H. INBOARD FAN DOOR
UNLOCK
R.H. OUT BOARD FAN DOOR
NO. 2 A/C
FLIGHT MODE CONVERSION
CTOL TO VTOL
GROUND POWER, BOTH HYD. SYSTEMS
22 FEB. 64
4 SEC.
TIME

FIGURE- 6.3.B4

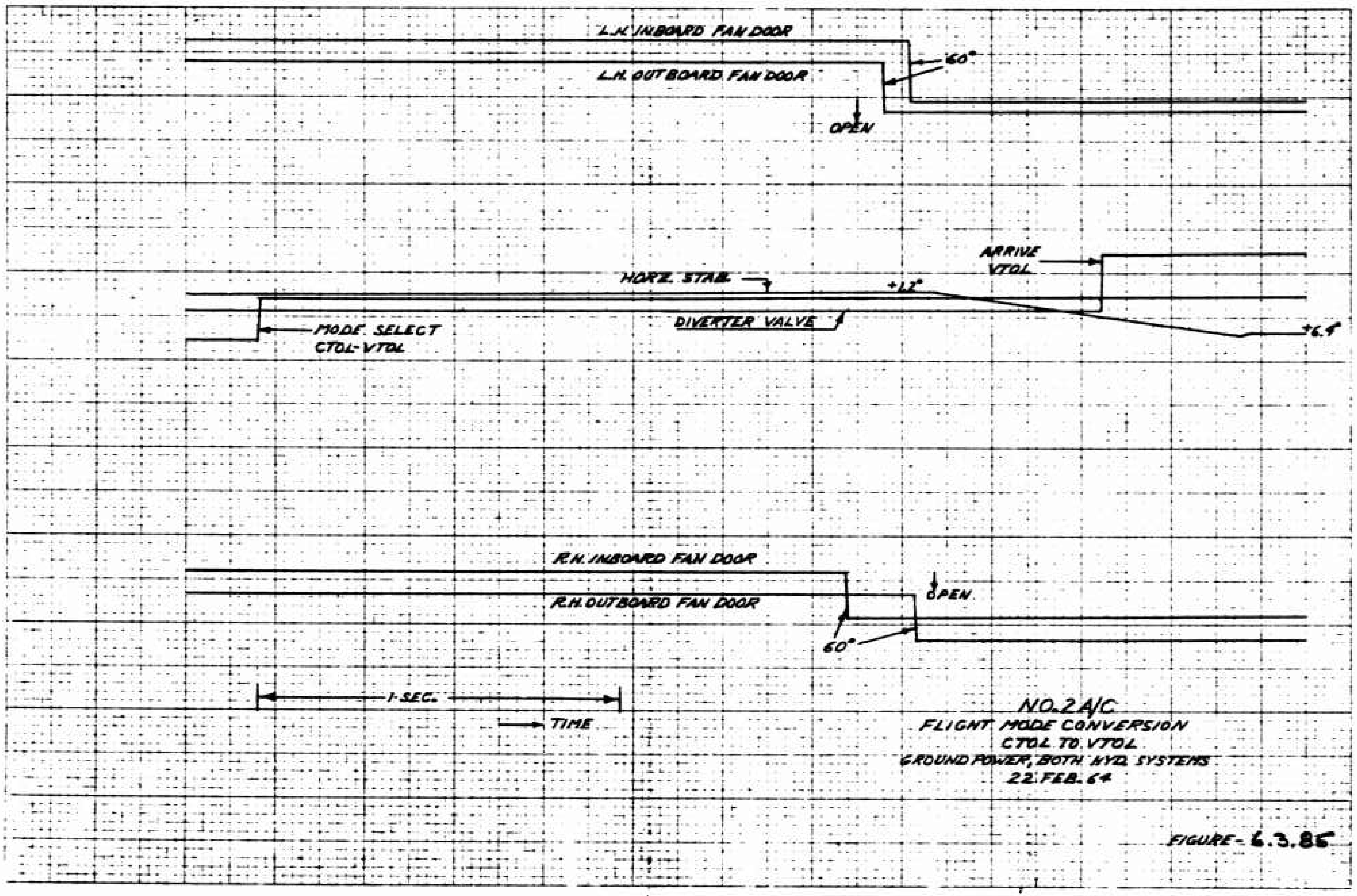

L.H. INBOARD FAN DOOR
L.H. OUTBOARD FAN DOOR
60°
OPEN
ARRIVE VTOL
HORZ. STAB.
+12°
DIVERTER VALVE
MODE SELECT CTOL-VTOL
+6.4°
R.H. INBOARD FAN DOOR
R.H. OUTBOARD FAN DOOR
OPEN
60°
1-SEC.
TIME
NO. 2 A/C
FLIGHT MODE CONVERSION
CTOL TO VTOL
GROUND POWER, BOTH HYD. SYSTEMS
22 FEB. 64

FIGURE - 6.3.85

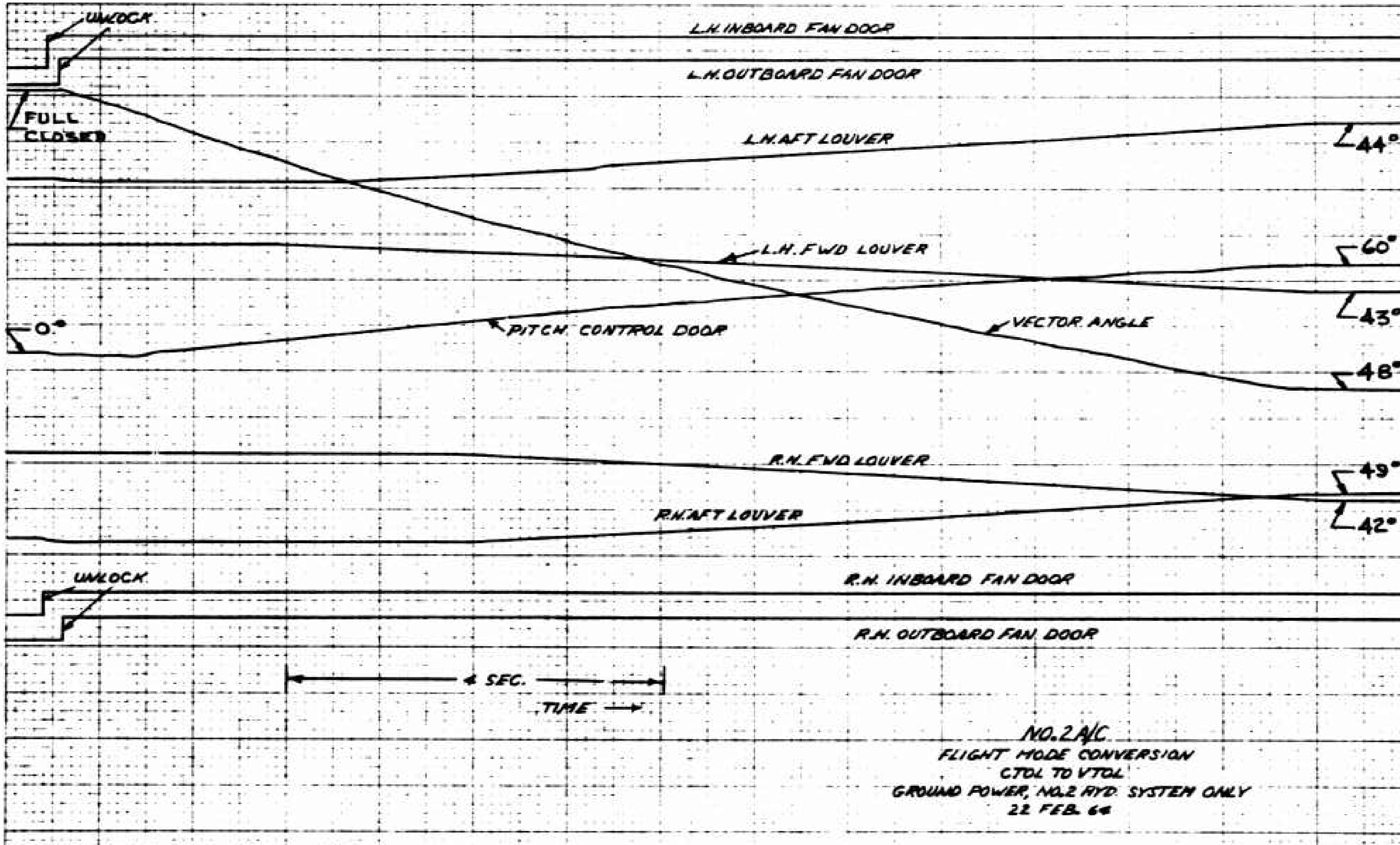
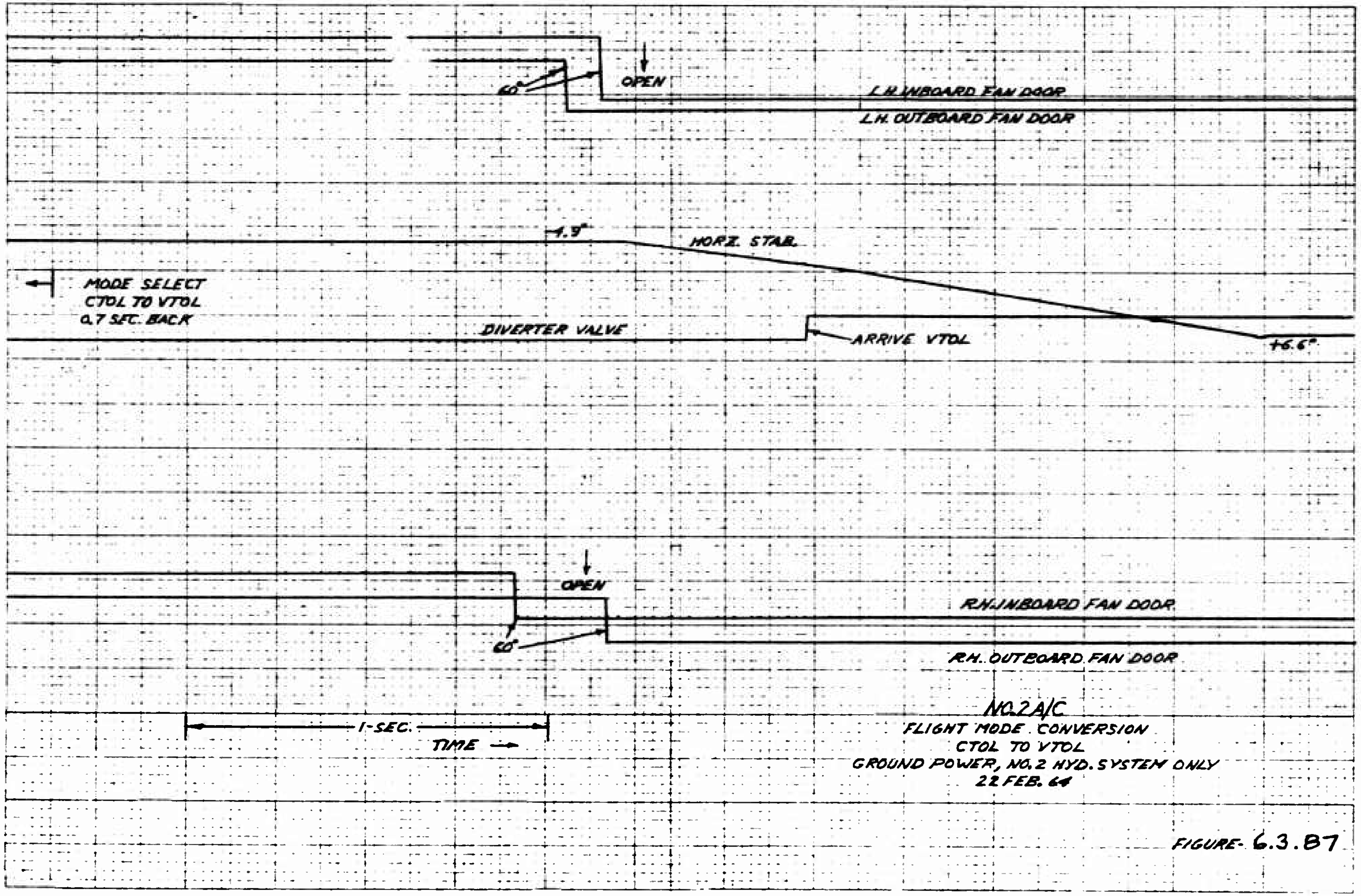

UNLOCK
L.H. INBOARD FAN DOOR
L.H. OUTBOARD FAN DOOR
FULL CLOSED
L.H. AFT LOUVER
44°
L.H. FWD LOUVER
60°
43°
0°
PITCH CONTROL DOOR
VECTOR ANGLE
48°
R.H. FWD LOUVER
49°
R.H. AFT LOUVER
42°
UNLOCK
R.H. INBOARD FAN DOOR
R.H. OUTBOARD FAN DOOR
4 SEC.
TIME
NO. 2 A/C
FLIGHT MODE CONVERSION
CTOL TO VTOL
GROUND POWER, NO. 2 HYD. SYSTEM ONLY
22 FEB. 64

FIGURE- 6.3.86



NO. 2 A/C
FLIGHT MODE CONVERSION
CTOL TO VTOL
GROUND POWER, NO. 2 HYD. SYSTEM ONLY
22 FEB. 64

FIGURE 6.3.B7

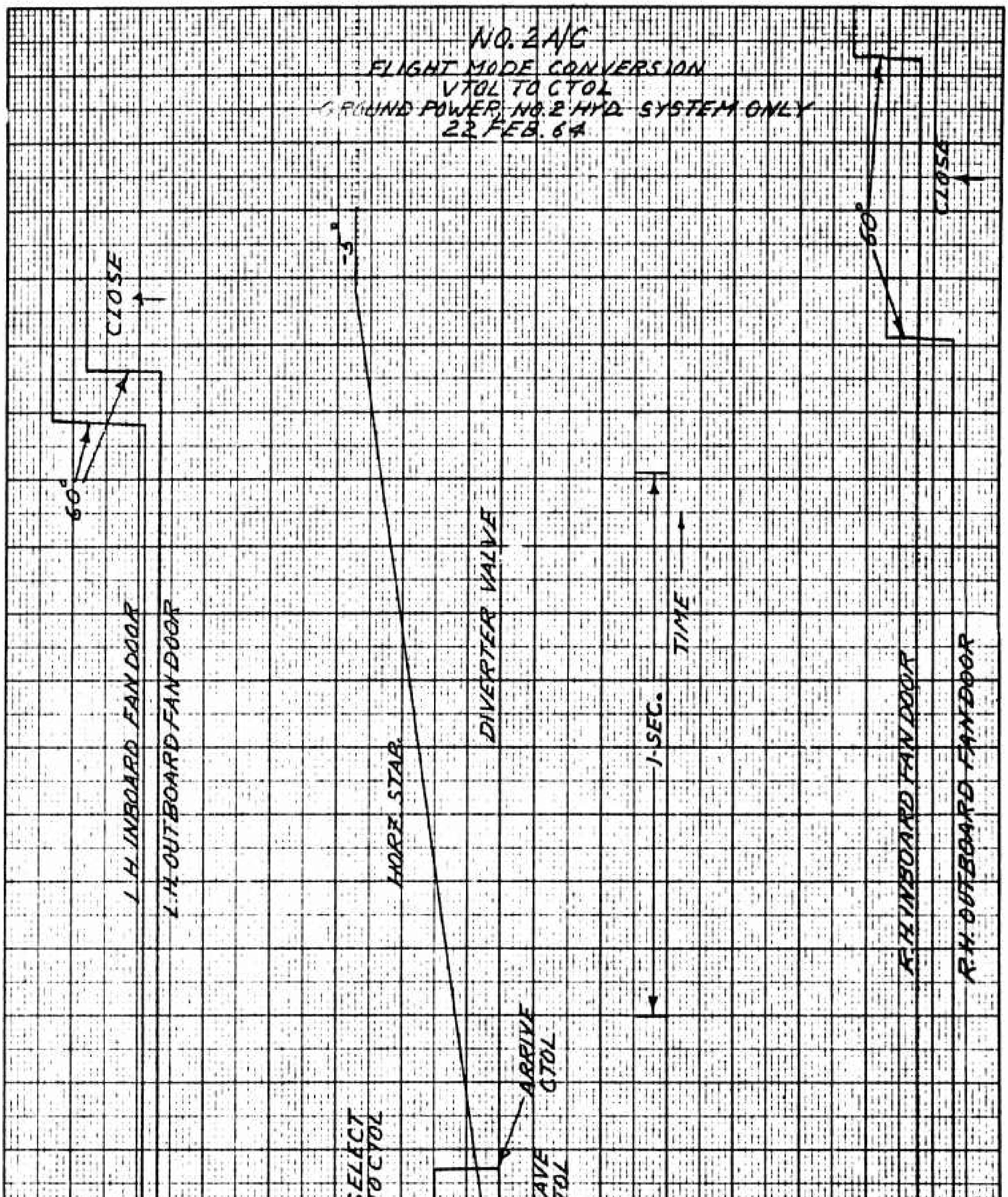
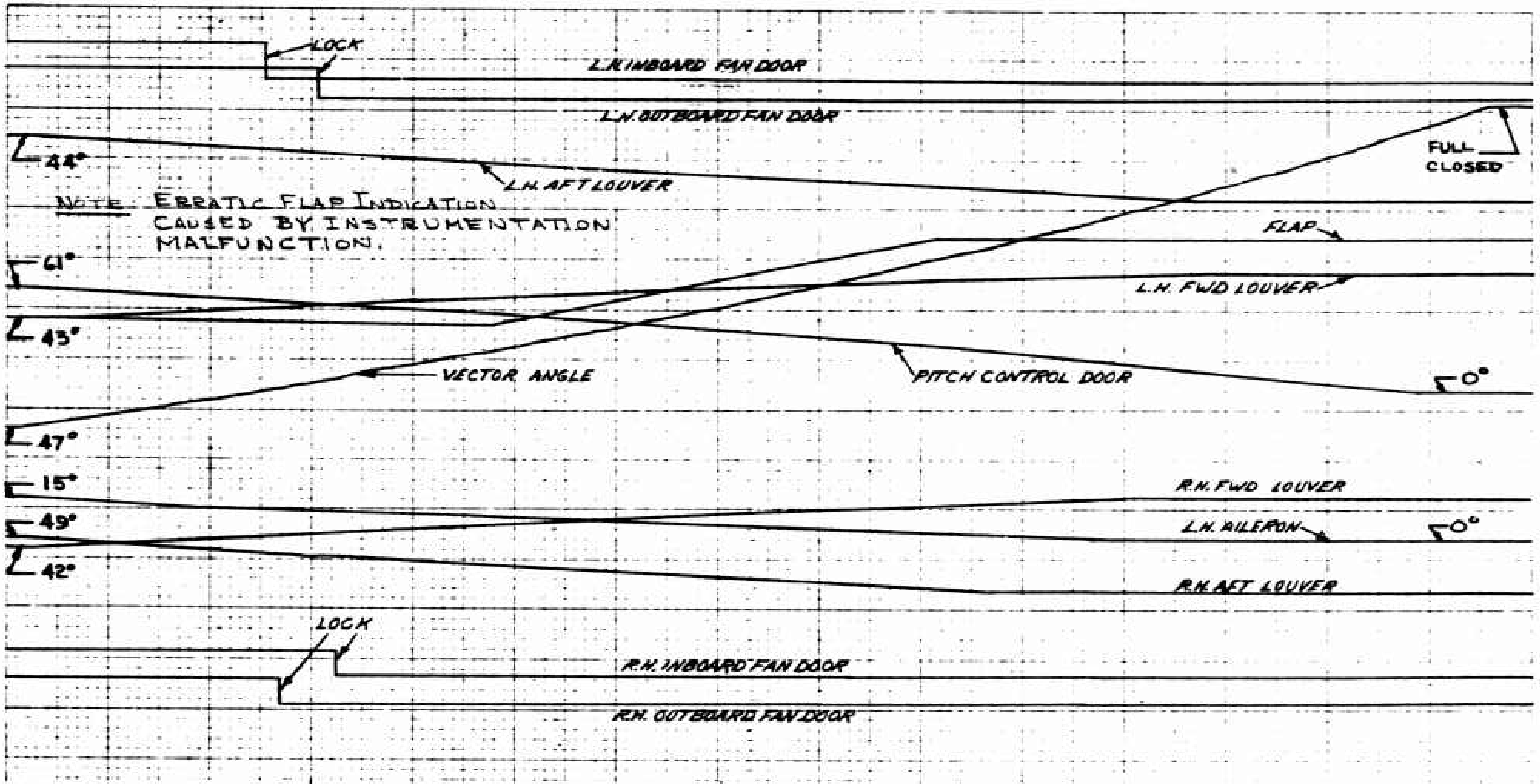
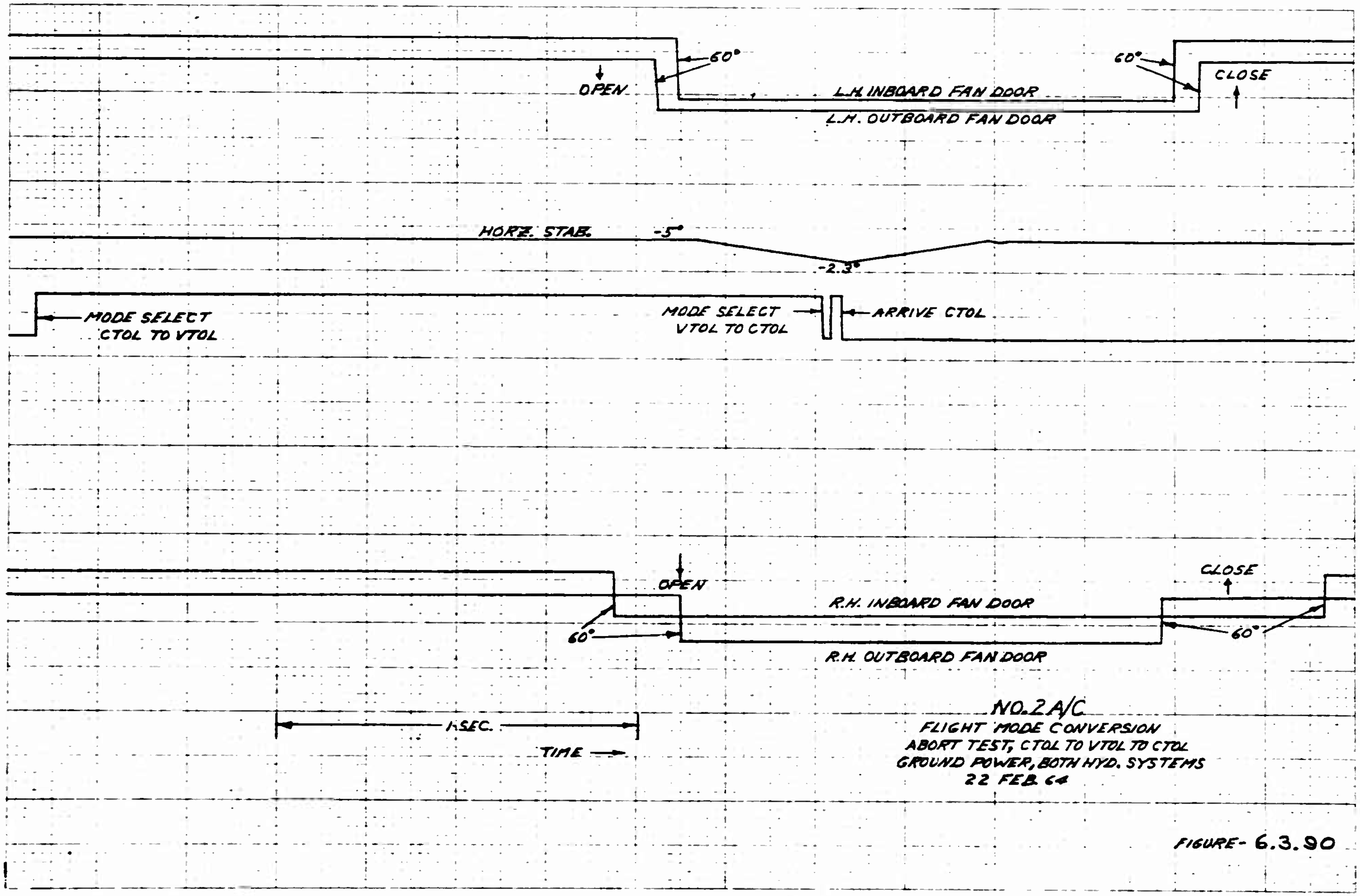

NO. 2 A/C
FLIGHT MODE CONVERSION
VTOL TO CTOL
GROUND POWER, NO. 2 HYD. SYSTEM ONLY
22 FEB. 64
L.H. INBOARD FAN DOOR
L.H. OUTBOARD FAN DOOR
CLOSE
60°
HORZ. STAB.
-3°
+6.9°
DIVERTER VALVE
ARRIVE CTOL
LEAVE VTOL
MODE SELECT VTOL TO CTOL
1-SEC.
TIME
R.H. INBOARD FAN DOOR
R.H. OUTBOARD FAN DOOR
60°
CLOSE

FIGURE- 6.3.88



NO. 2 A/C
FLIGHT MODE CONVERSION
VTOL TO CTOL
GROUND POWER, NO. 2 HYD. SYSTEM ONLY
22 FEB. 64

FIGURE- 6.3.B9



NO. 2 A/C
FLIGHT MODE CONVERSION
ABORT TEST, CTOL TO VTOL TO CTOL
GROUND POWER, BOTH HYD. SYSTEMS
22 FEB 64

FIGURE- 6.3.90

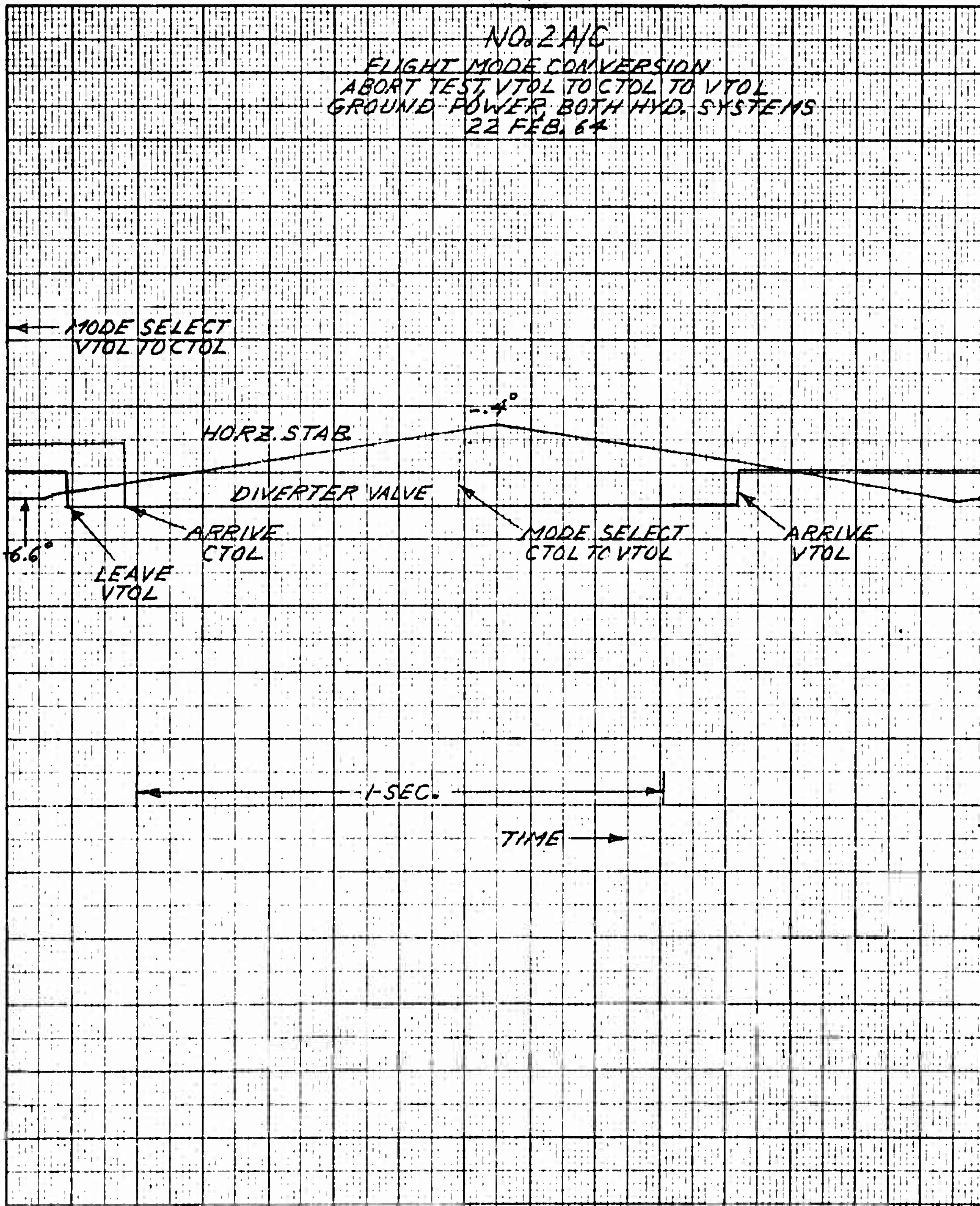
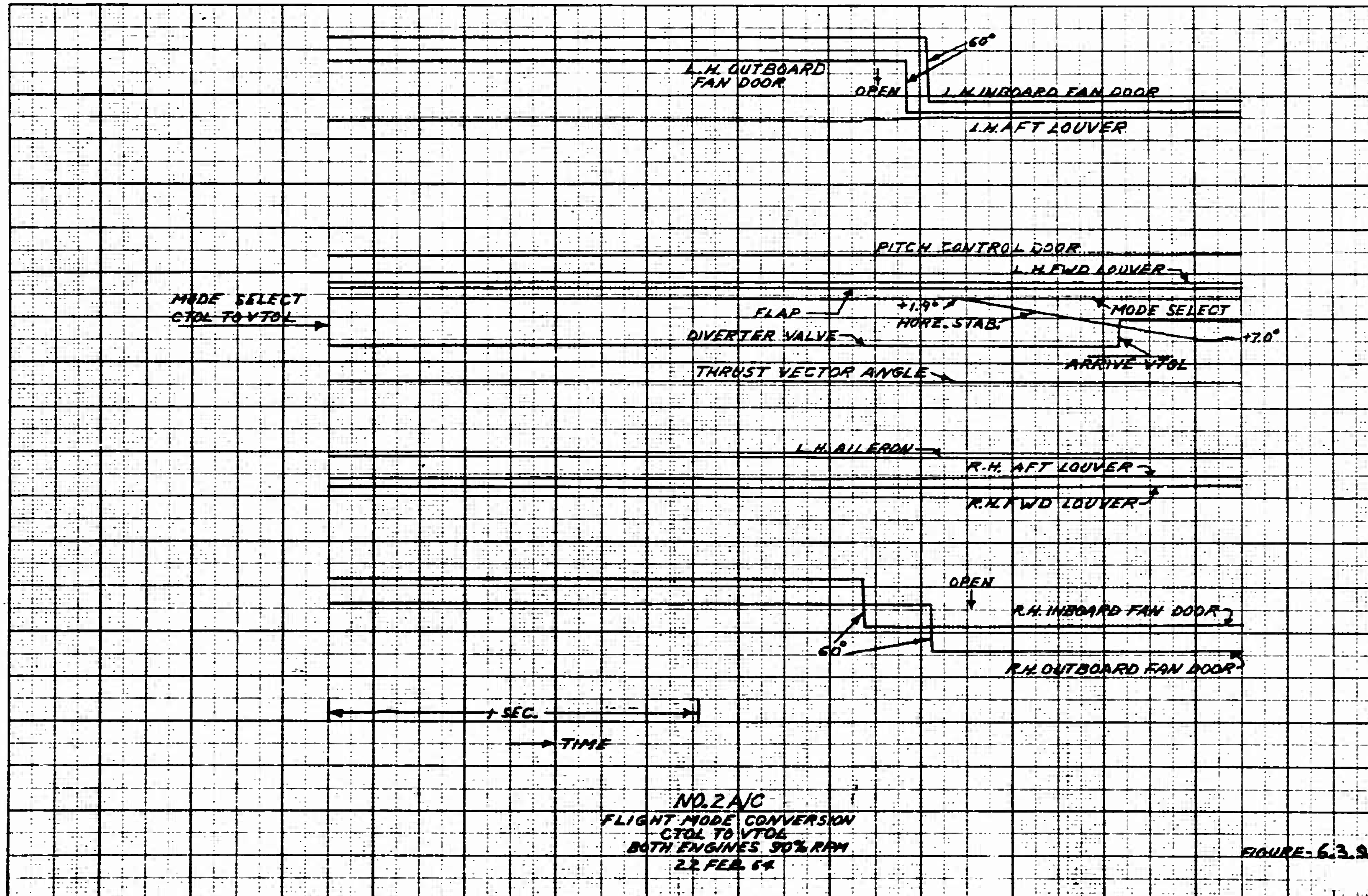
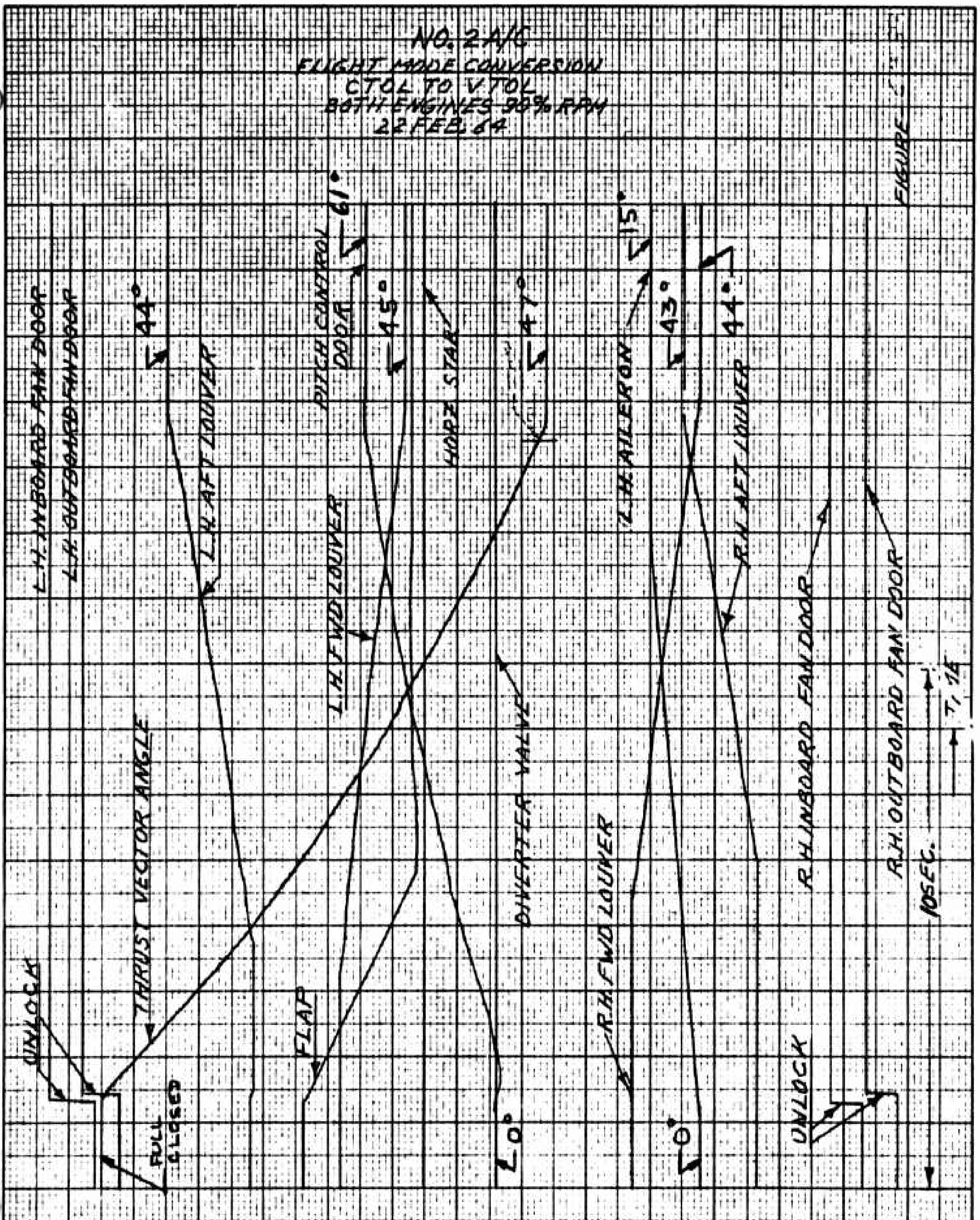

NO. 2 A/C
FLIGHT MODE CONVERSION
ABORT TEST VTOL TO CTOL TO VTOL
GROUND POWER, BOTH HYD. SYSTEMS
22 FEB. 64
MODE SELECT
VTOL TO CTOL
-.4°
HORZ. STAB.
DIVERTER VALVE
+6.6°
LEAVE
VTOL
ARRIVE
CTOL
MODE SELECT
CTOL TO VTOL
ARRIVE
VTOL
1-SEC.
TIME

FIGURE-6.3.91



NO. 2 A/C
FLIGHT MODE CONVERSION
CTOL TO VTOL
BOTH ENGINES 90% RPM
22 FEB. 64

FIGURE-6.3.92



Note: Erratic Flap Indication
Caused by Instrumentation
Malfunction

FIGURE-6.3.93

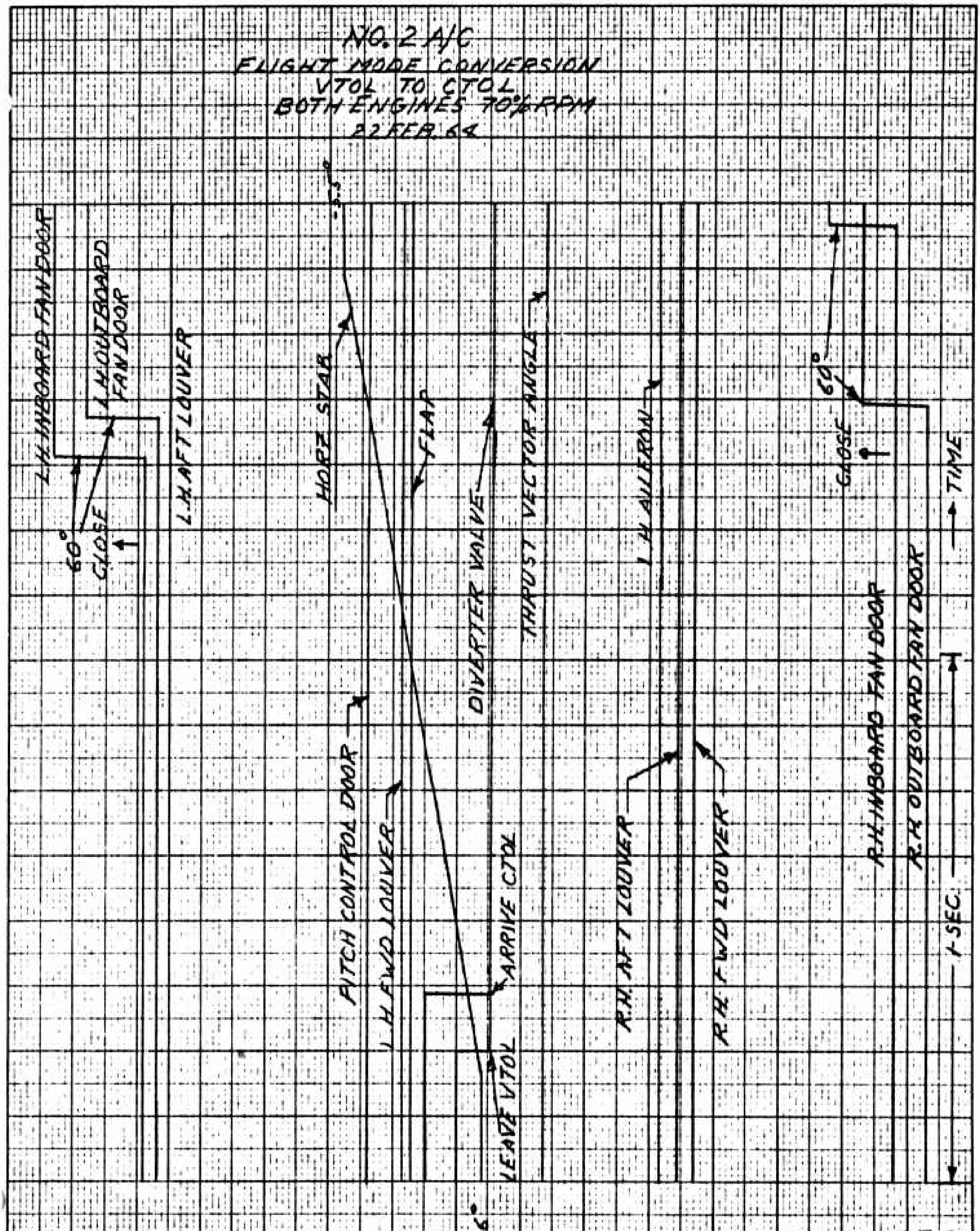
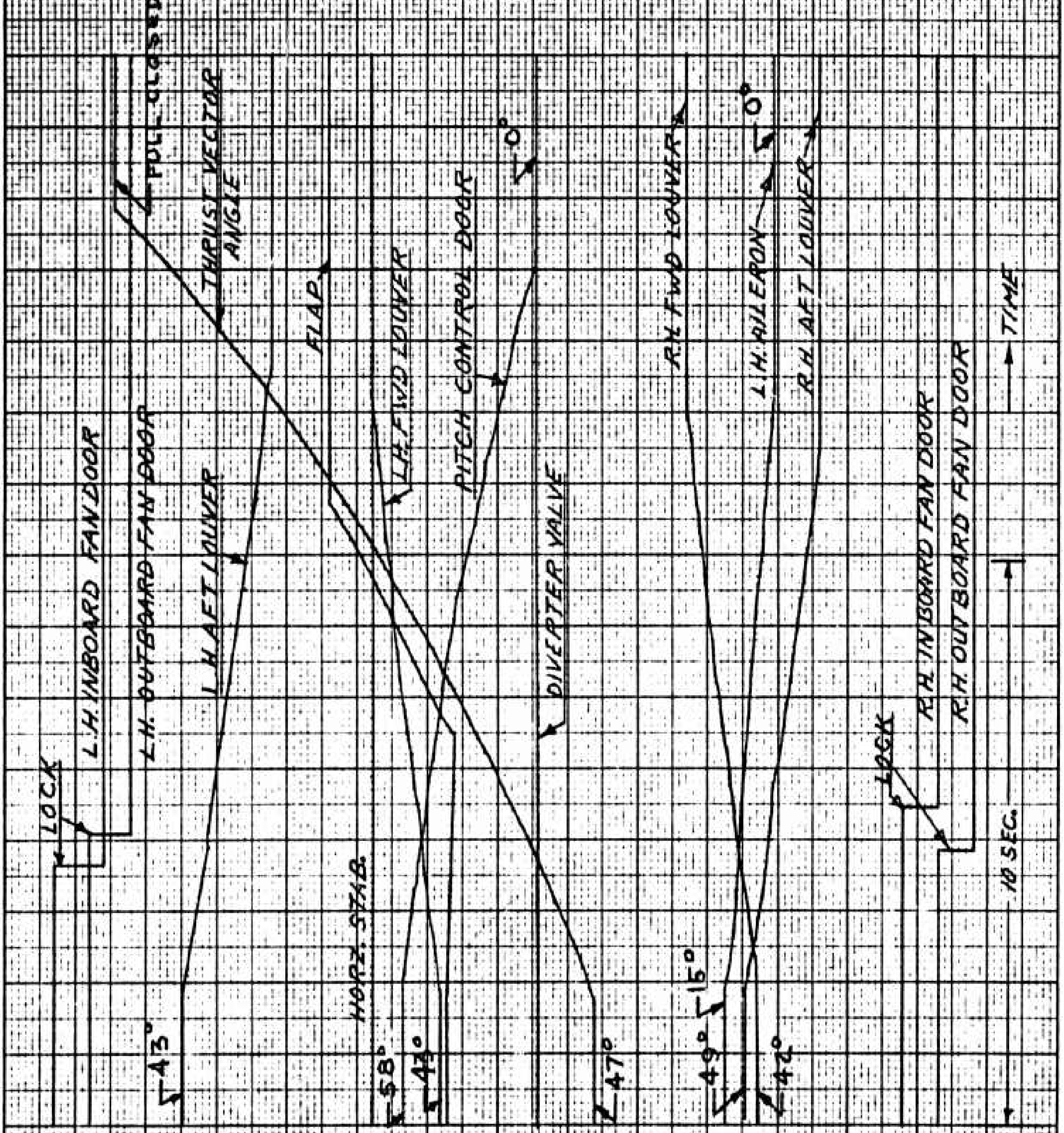

NO. 2 A/C
FLIGHT MODE CONVERSION
VTOL TO CTOL
BOTH ENGINES 70% RPM
22 FEB. 64
L.H. INBOARD FAN DOOR
L.H. OUTBOARD FAN DOOR
60°
CLOSE
L.H. AFT LOUVER
-3.5°
HORZ STAB
PITCH CONTROL DOOR
L.H. FWD LOUVER
FLAP
DIVERTER VALVE
ARRIVE CTOL
LEAVE VTOL
+6.5°
THRUST VECTOR ANGLE
L.H. AILERON
R.H. AFT LOUVER
R.H. FWD LOUVER
60°
CLOSE
R.H. INBOARD FAN DOOR
R.H. OUTBOARD FAN DOOR
TIME
1-SEC.

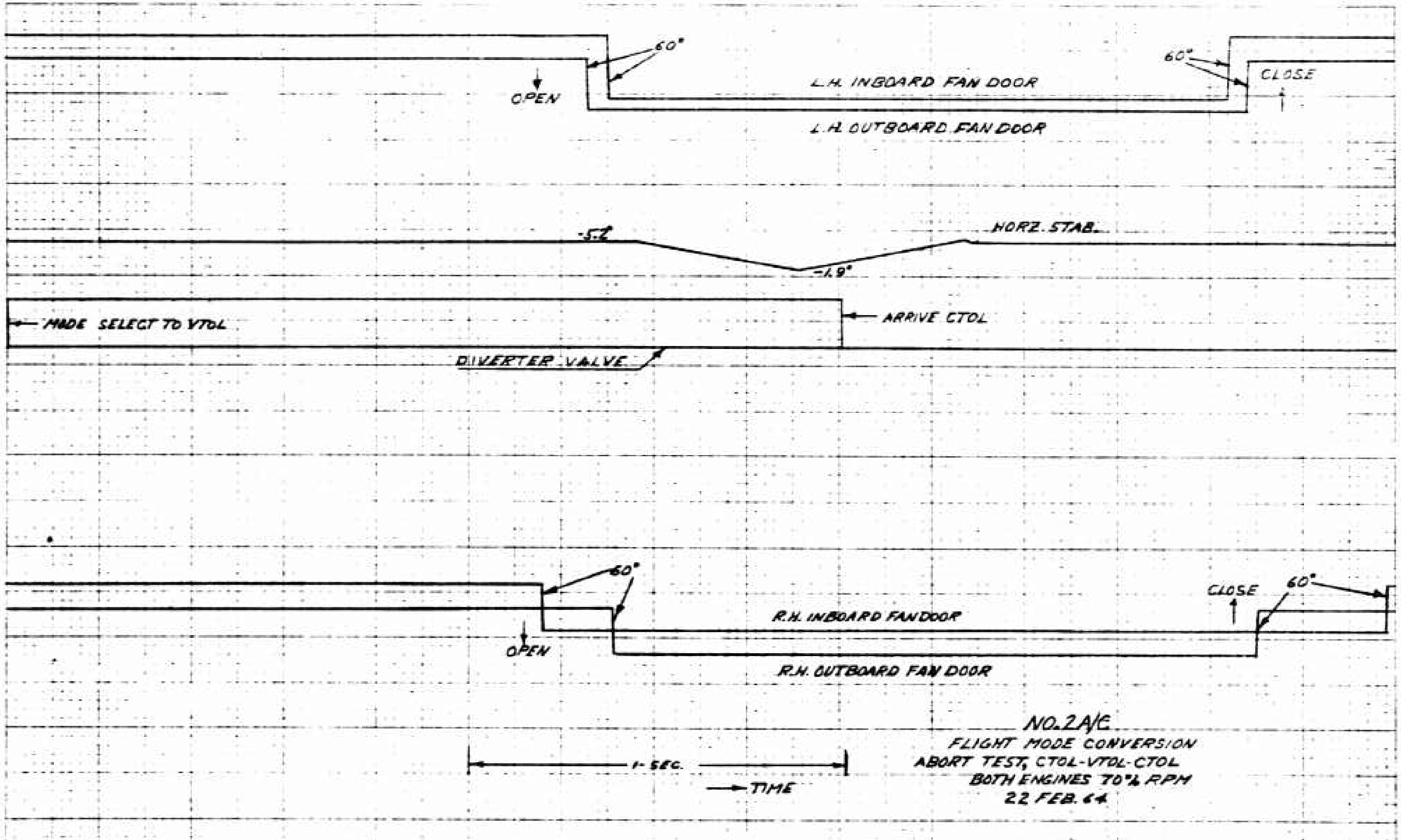
FIGURE - 6.3.94

NO. 2 A/C
FLIGHT MODE CONVERSION
VTOL TO CTOL
BOTH ENGINES 70% RPM
22 FEB. 64



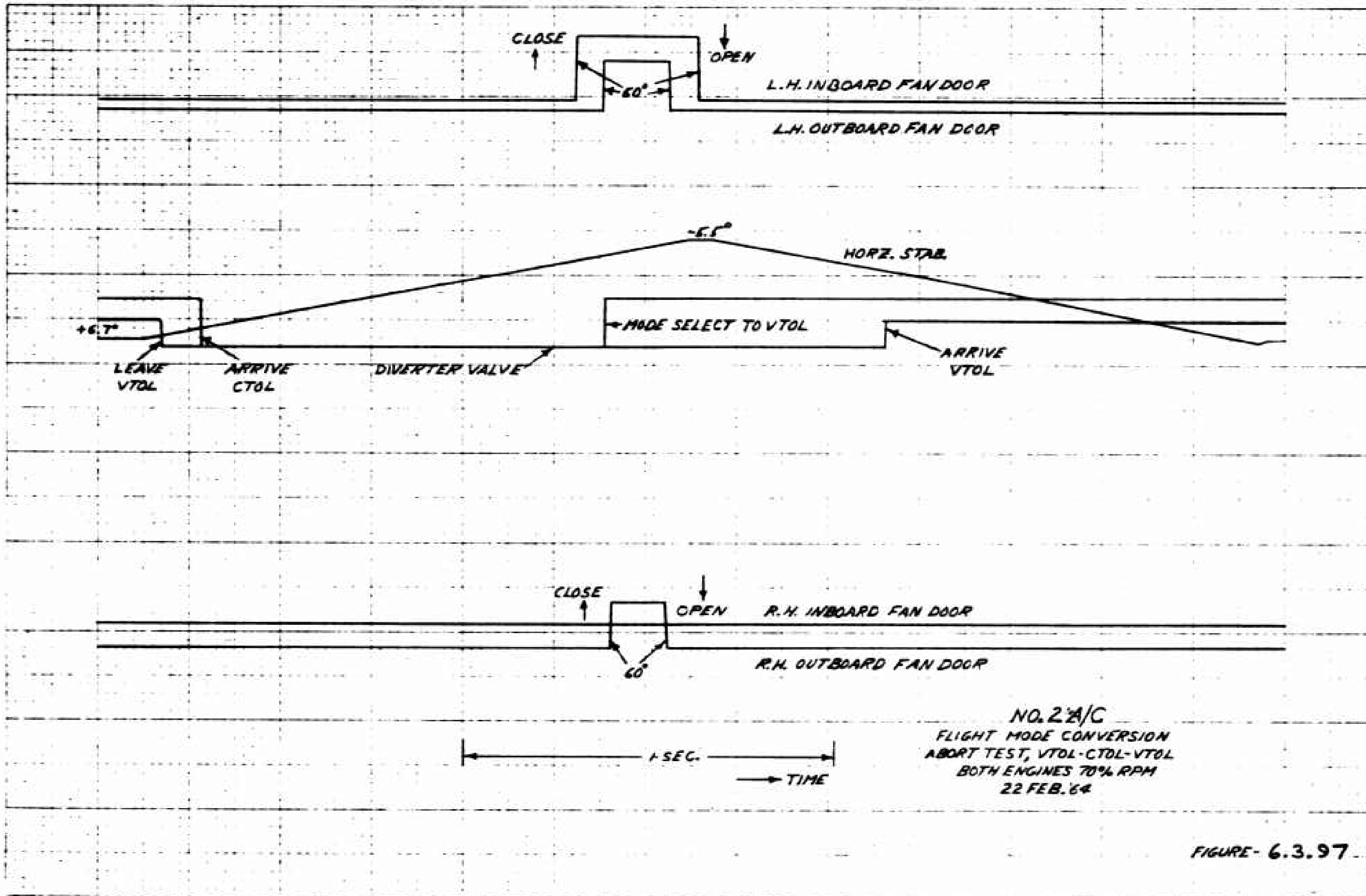
NOTE: ERRATIC FLAP INDICATION CAUSED BY INSTRUMENTATION MALFUNCTION.

FIGURE-6.3.95



60°
OPEN
L.H. INBOARD FAN DOOR
60°
CLOSE
L.H. OUTBOARD FAN DOOR
-5.2
HORZ. STAB.
-1.9°
MODE SELECT TO VTOL
ARRIVE CTOL
DIVERTER VALVE
60°
CLOSE
60°
R.H. INBOARD FAN DOOR
OPEN
R.H. OUTBOARD FAN DOOR
NO.2 A/C
FLIGHT MODE CONVERSION
ABORT TEST, CTOL-VTOL-CTOL
BOTH ENGINES 70% RPM
22 FEB. 64
1-SEC.
TIME

FIGURE- 6.3.96



NO. 2 A/C
FLIGHT MODE CONVERSION
ABORT TEST, VTOL-CTOL-VTOL
BOTH ENGINES 70% RPM
22 FEB. '64

FIGURE- 6.3.97

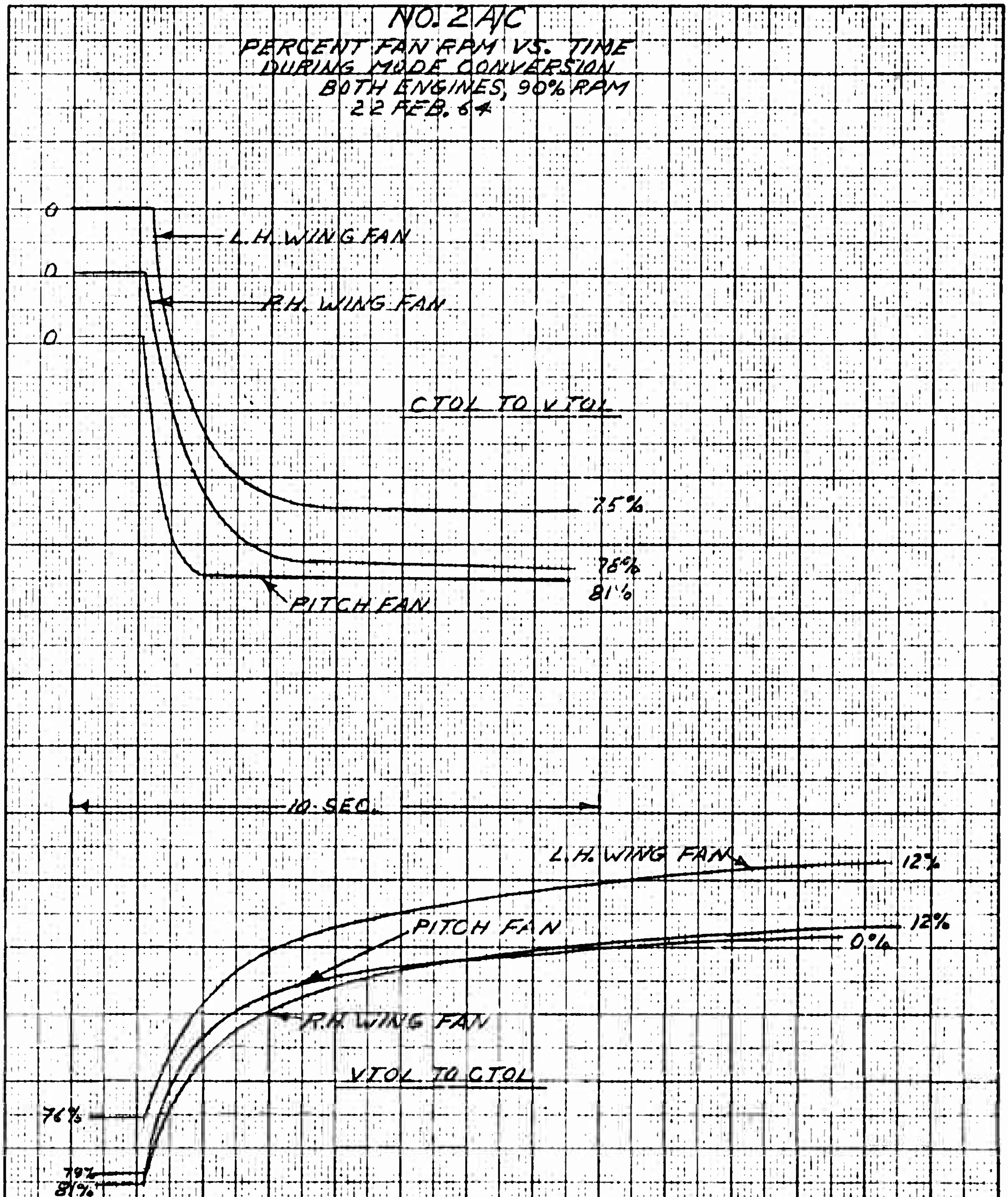
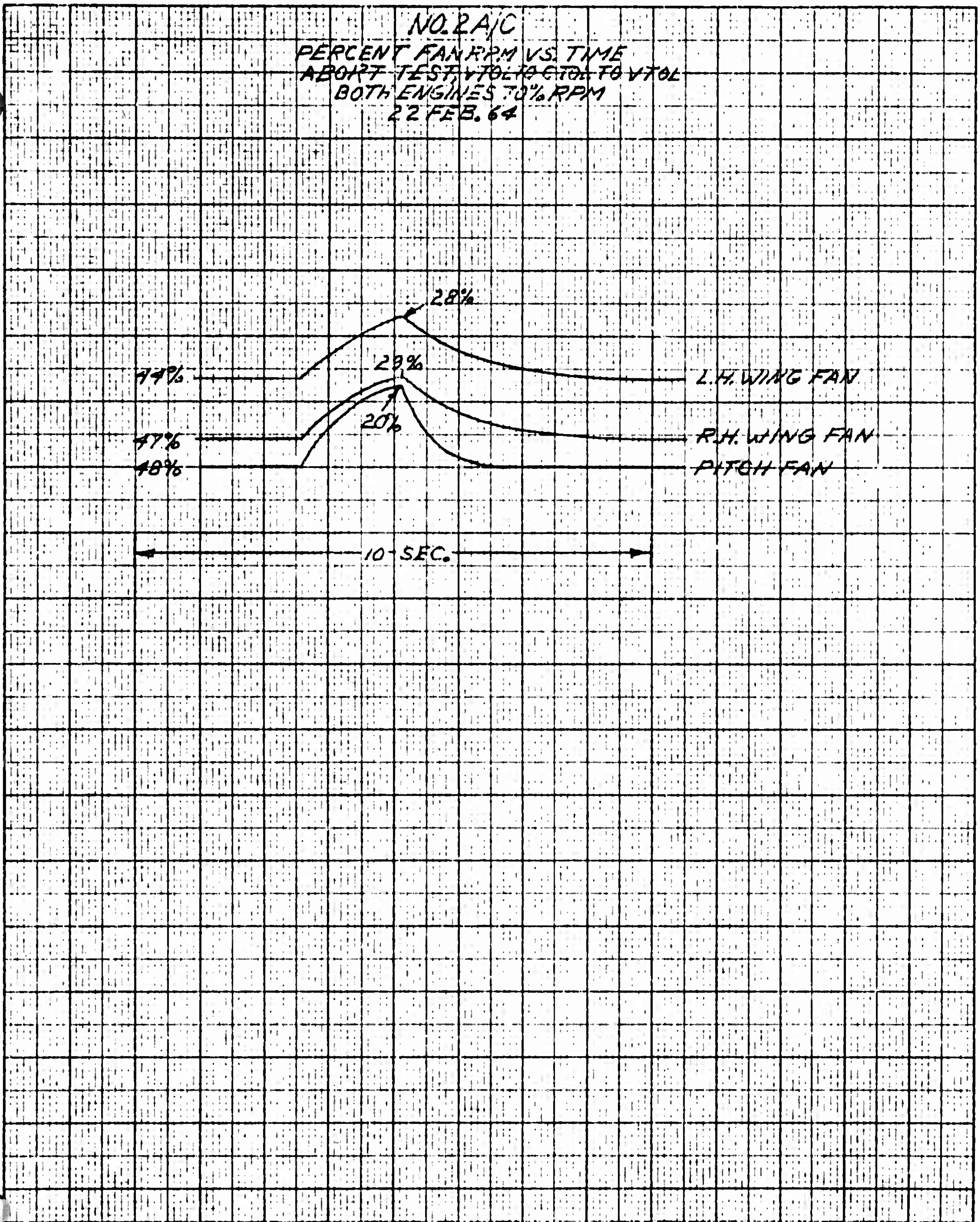
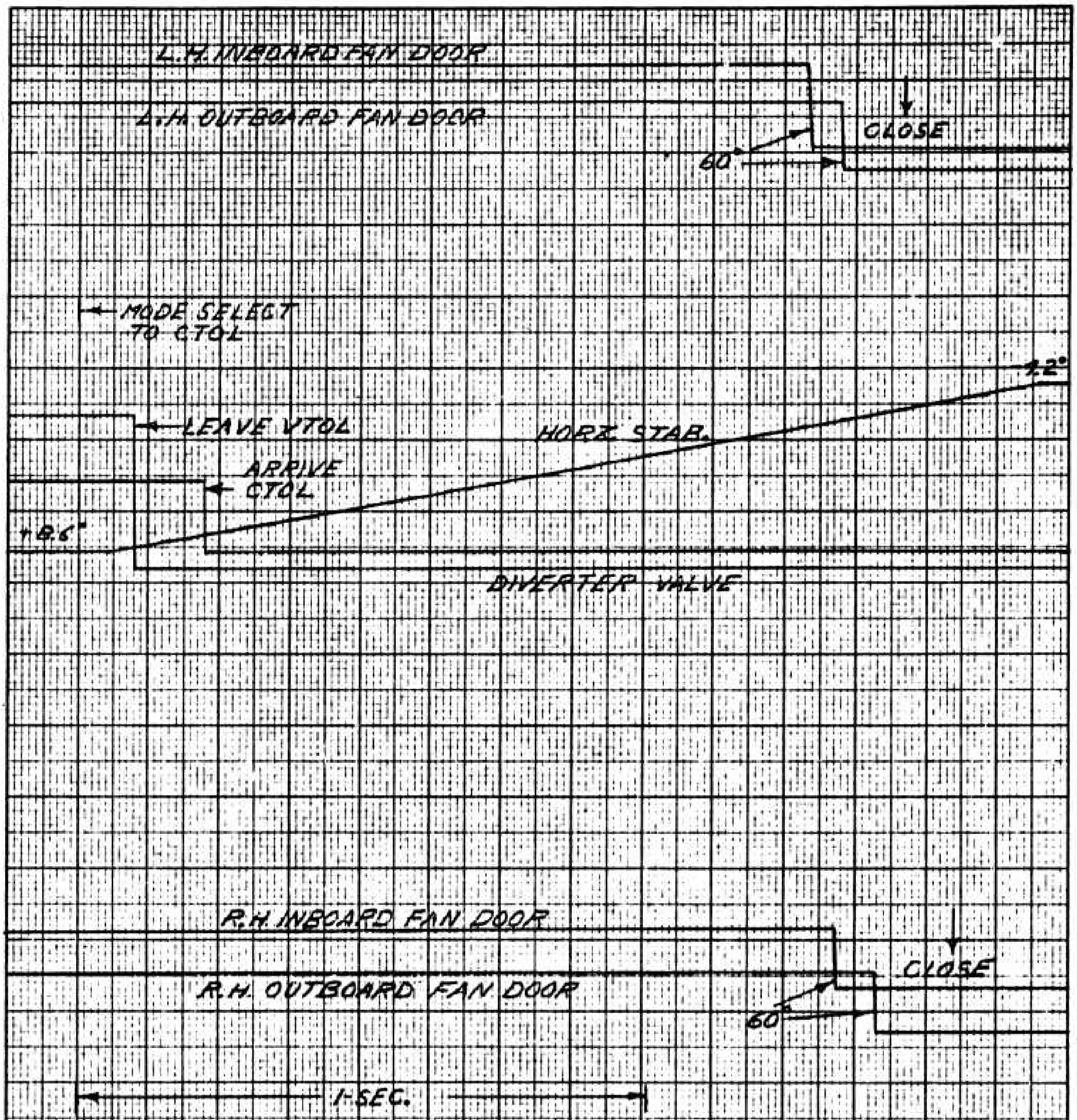

NO. 2 A/C
PERCENT FAN RPM VS. TIME
DURING MODE CONVERSION
BOTH ENGINES, 90% RPM
22 FEB. 64
L.H. WING FAN
R.H. WING FAN
CTOL TO VTOL
75%
78%
81%
PITCH FAN
10 SEC.
L.H. WING FAN
12%
PITCH FAN
12%
0%
R.H. WING FAN
VTOL TO CTOL
76%
79%
81%

FIGURE-6.3.98



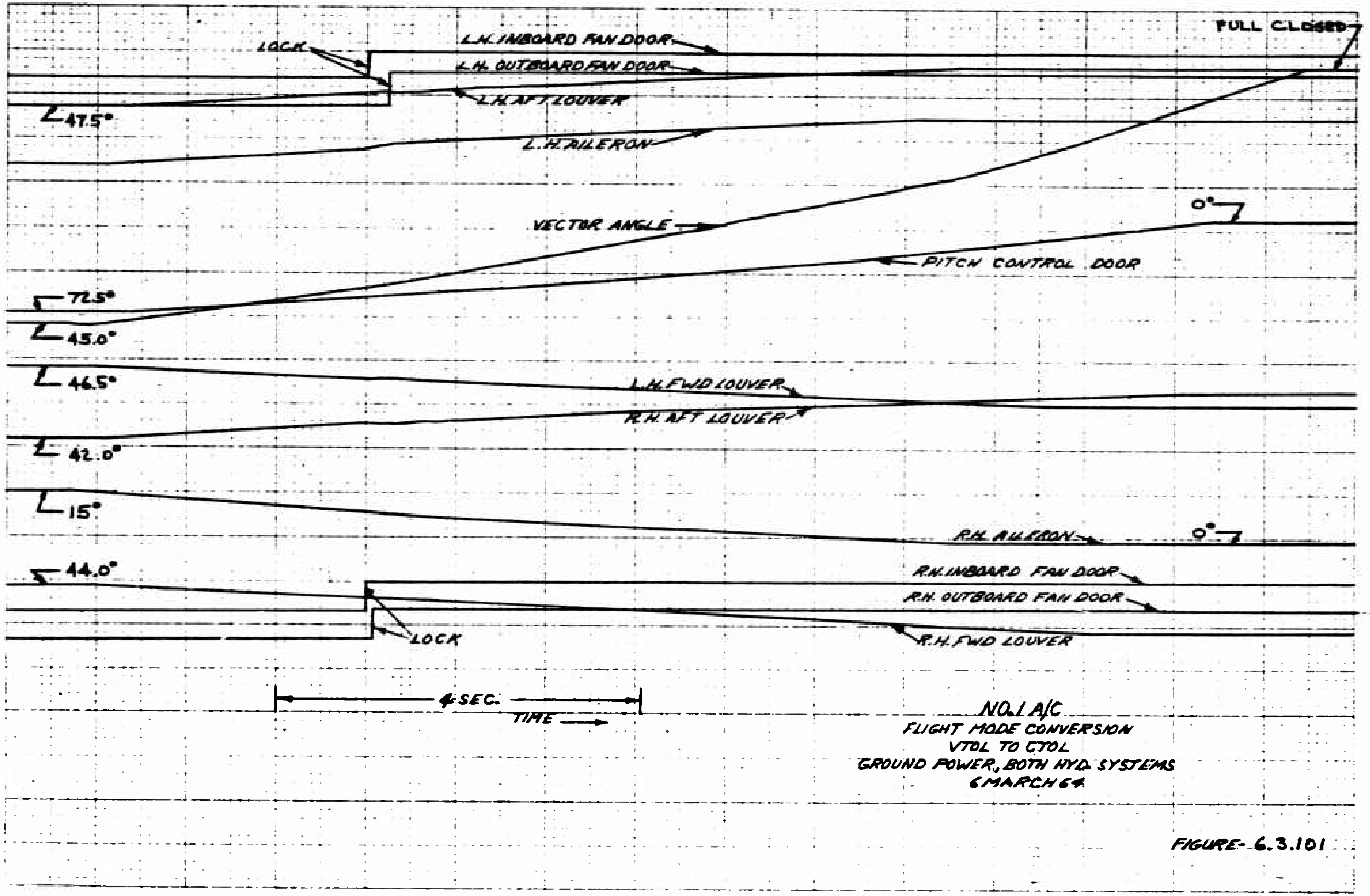
NO. 2 A/C
PERCENT FAN RPM VS. TIME
ABORT TEST, VTOL TO CTOL TO VTOL
BOTH ENGINES 70% RPM
22 FEB. 64
28%
29%
20%
44%
47%
48%
L.H. WING FAN
R.H. WING FAN
PITCH FAN
10 SEC.

FIGURE- 6.3.99



NO. 1 A/C
FLIGHT MODE CONVERSION
VTOL TO CTOL
GROUND POWER BOTH HYD. SYSTEMS
6 MARCH 64

FIGURE-63.100



NO. 1 A/C
FLIGHT MODE CONVERSION
VTOL TO CTOL
GROUND POWER, BOTH HYD. SYSTEMS
6 MARCH 64

FIGURE- 6.3.101

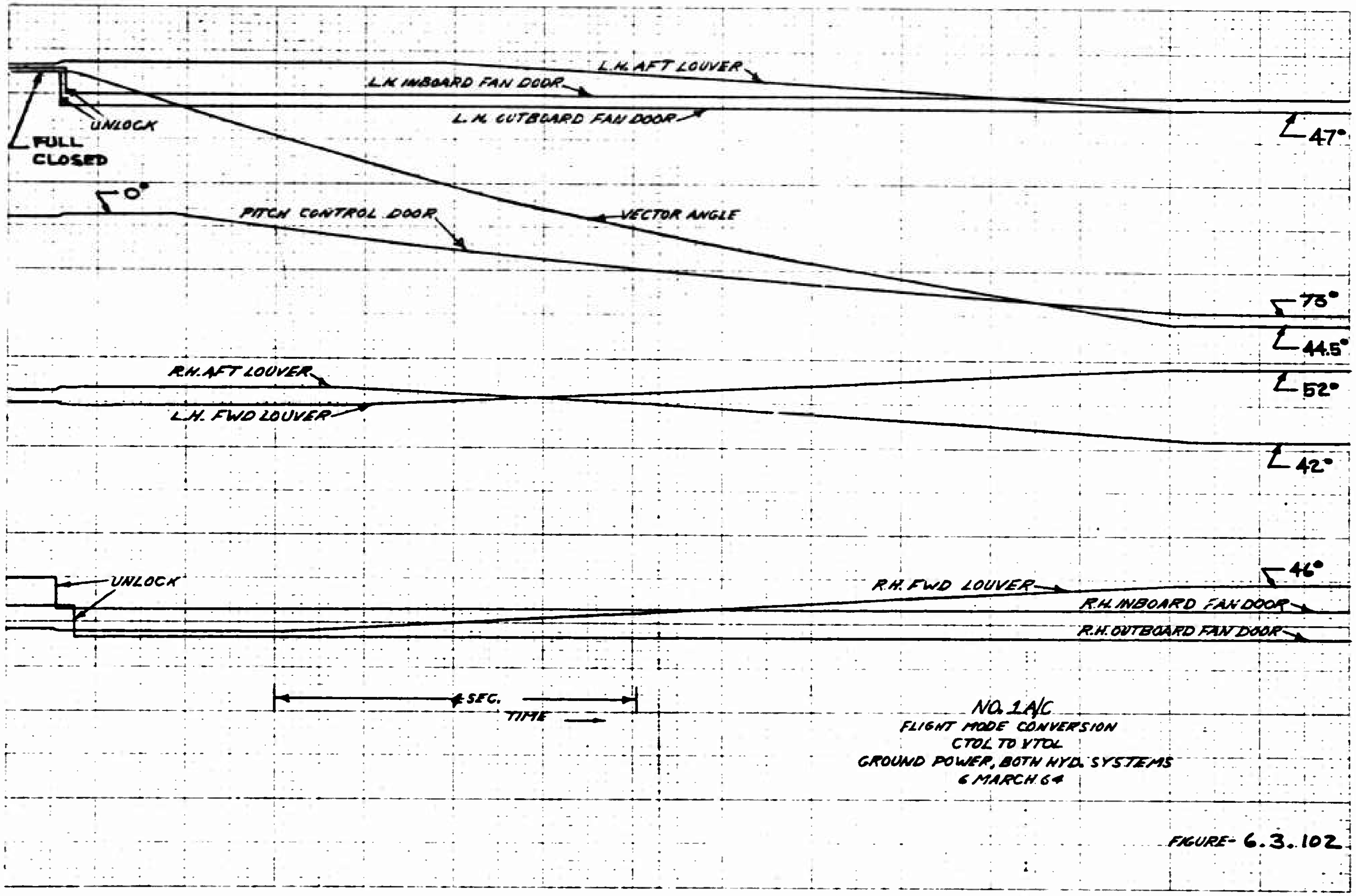


FIGURE- 6.3.102

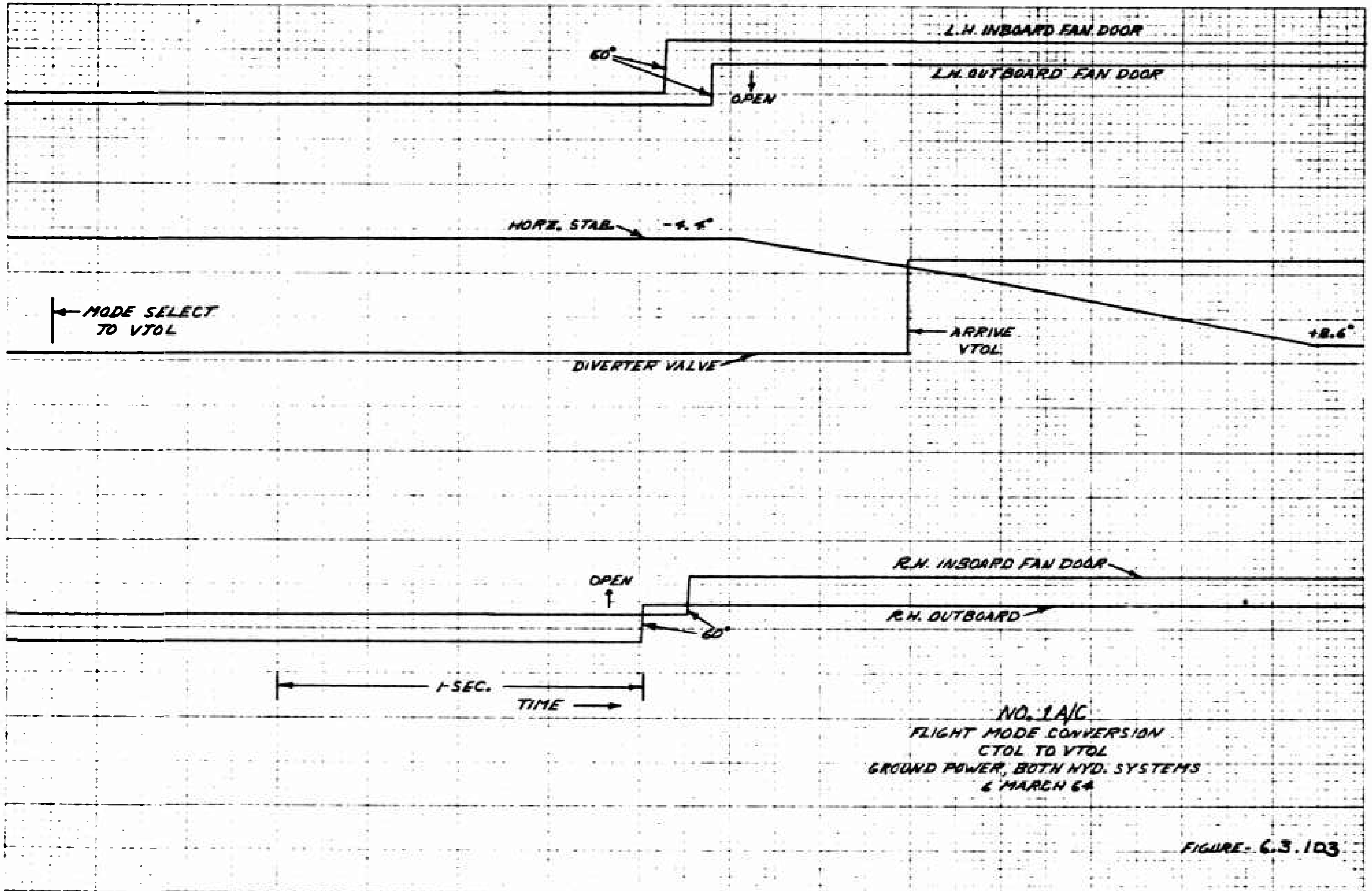
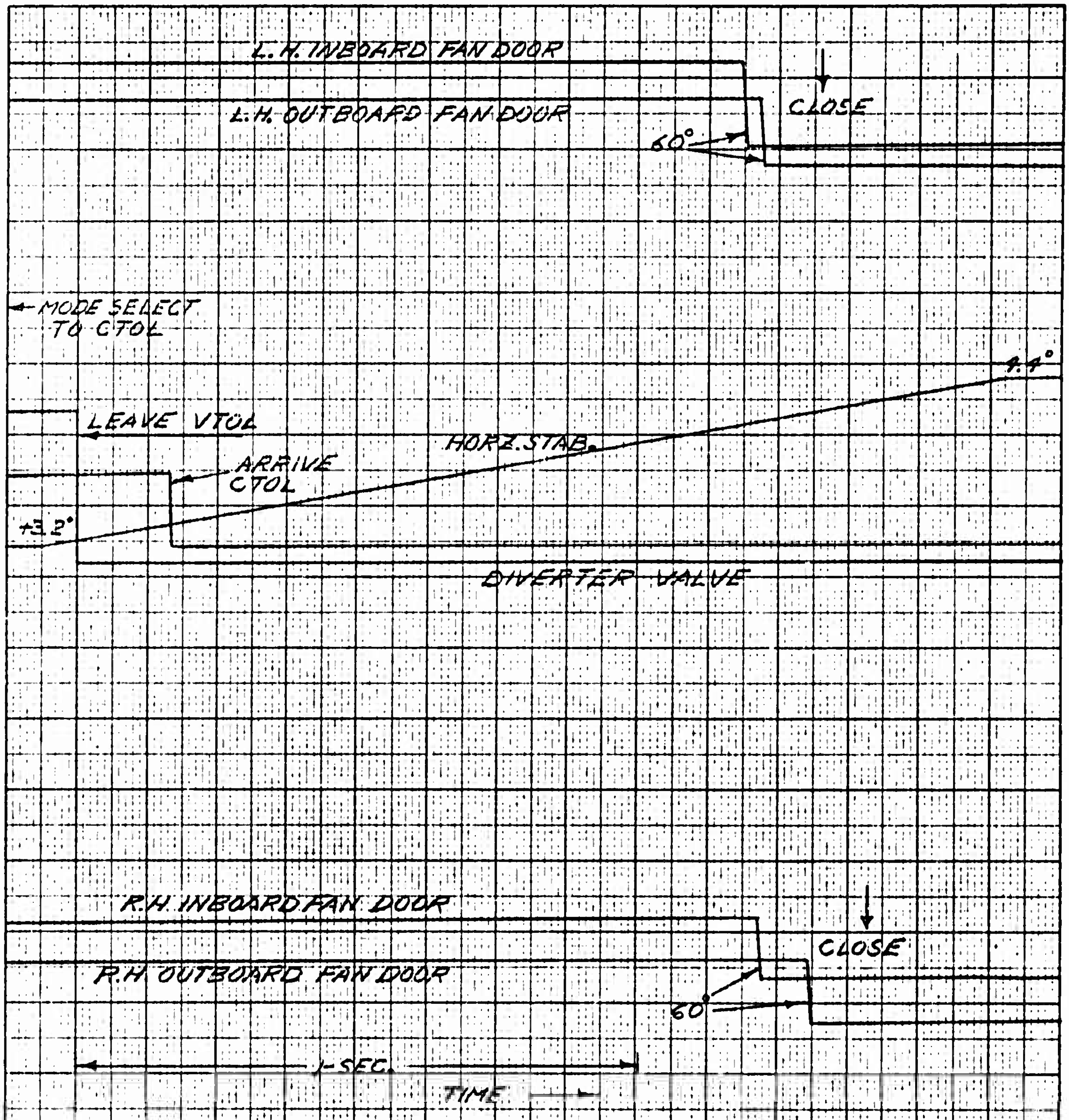


FIGURE- 6.3.103



NO. 1 A/C
FLIGHT MODE CONVERSION
VTOL TO CTOL
GROUND POWER, NO. 2 HYD. SYSTEM ONLY
6 MARCH 64

FIGURE 6.3.104

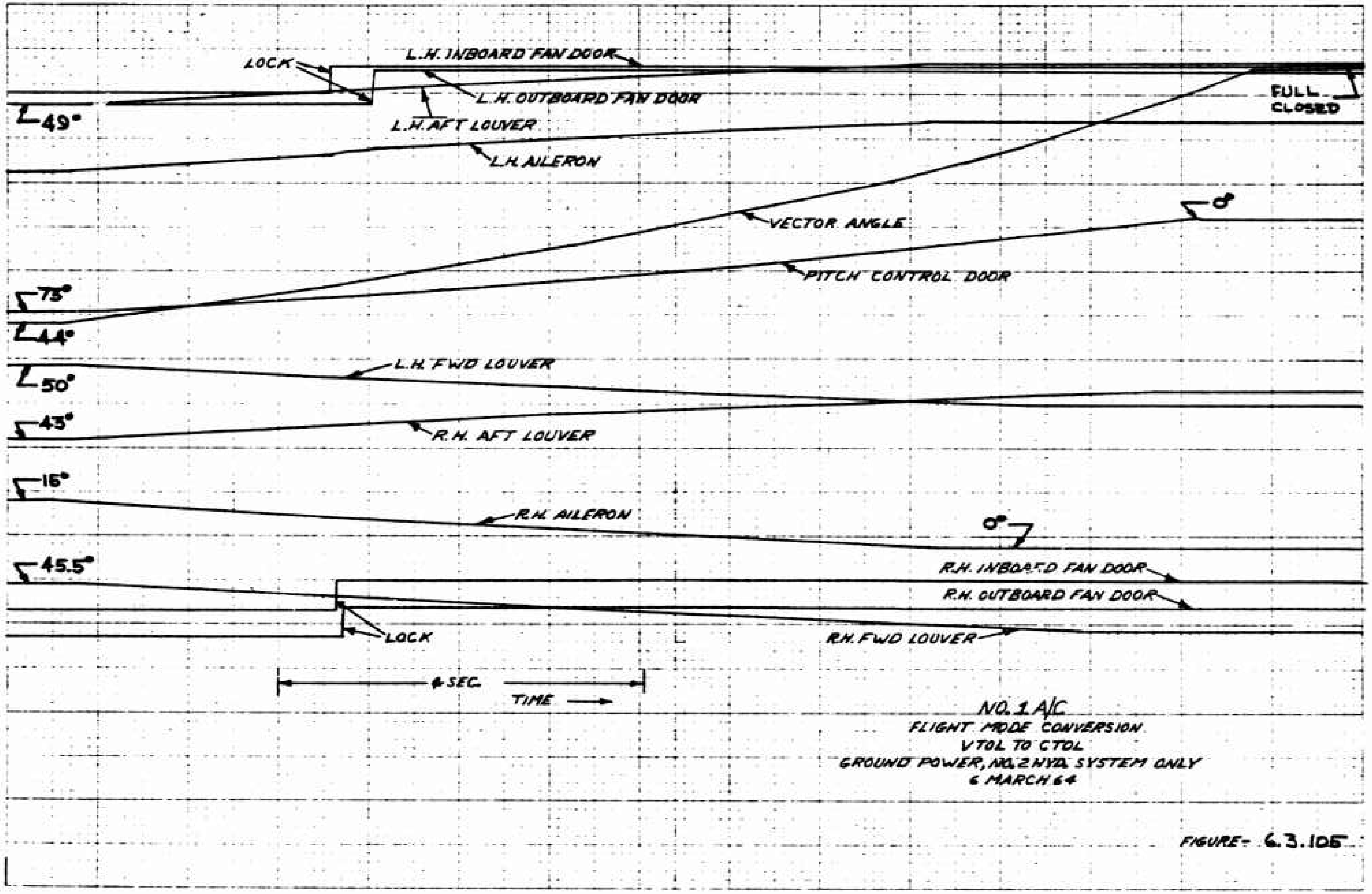


FIGURE- 6.3.105

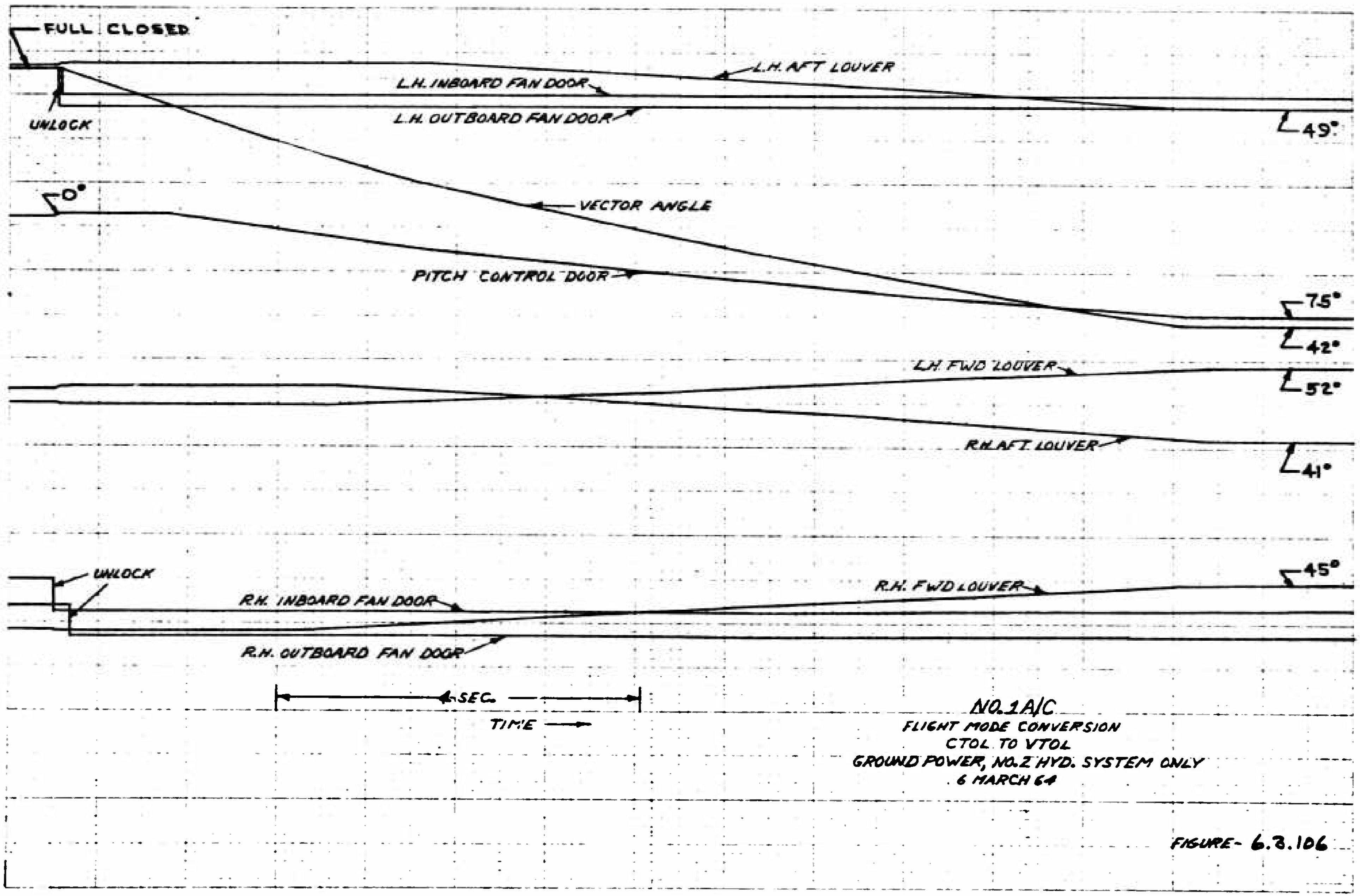
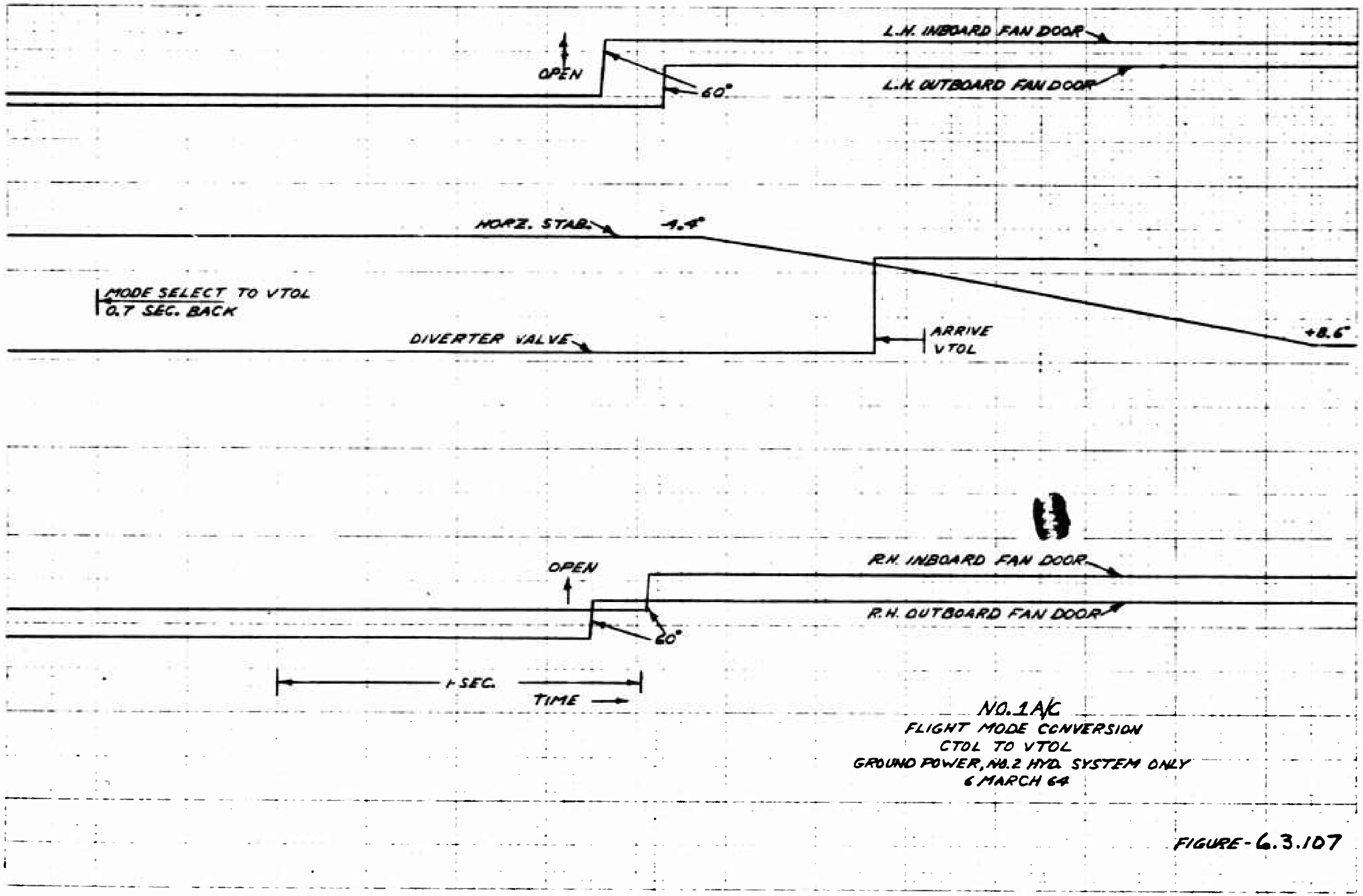
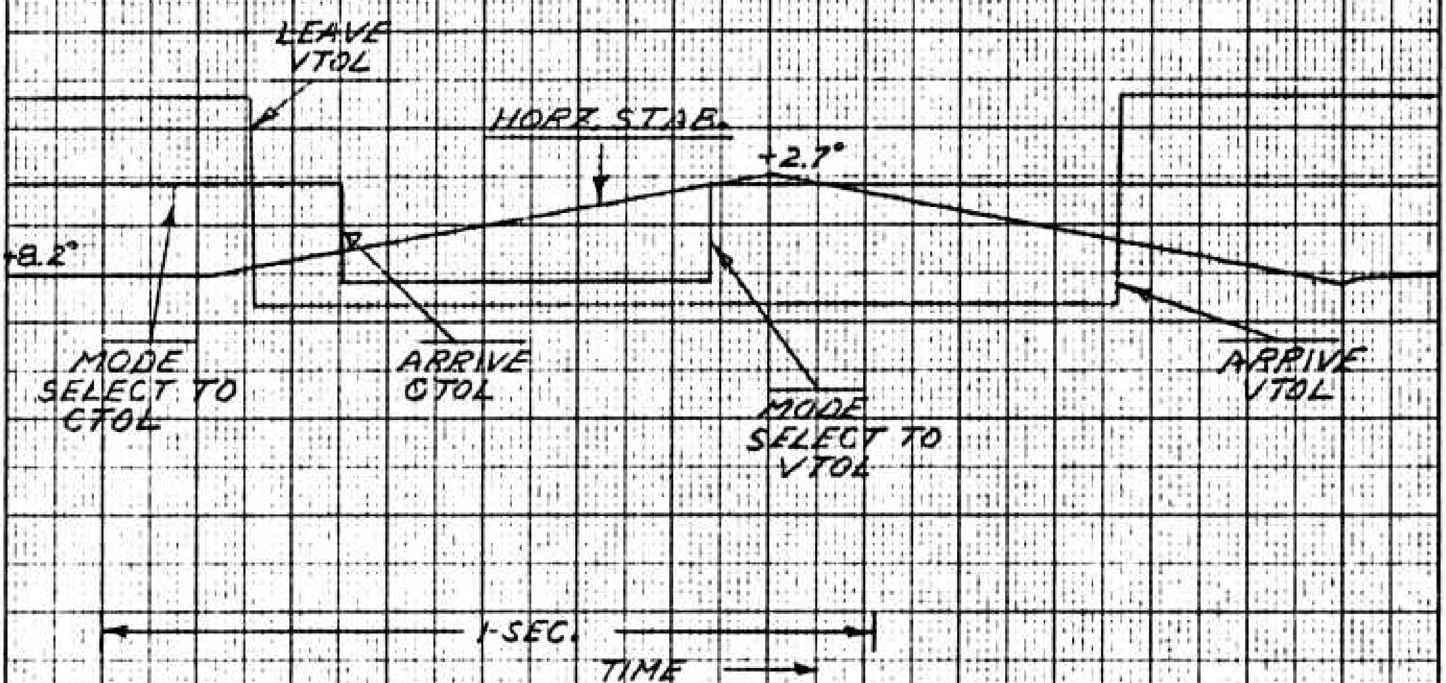


FIGURE- 6.3.106



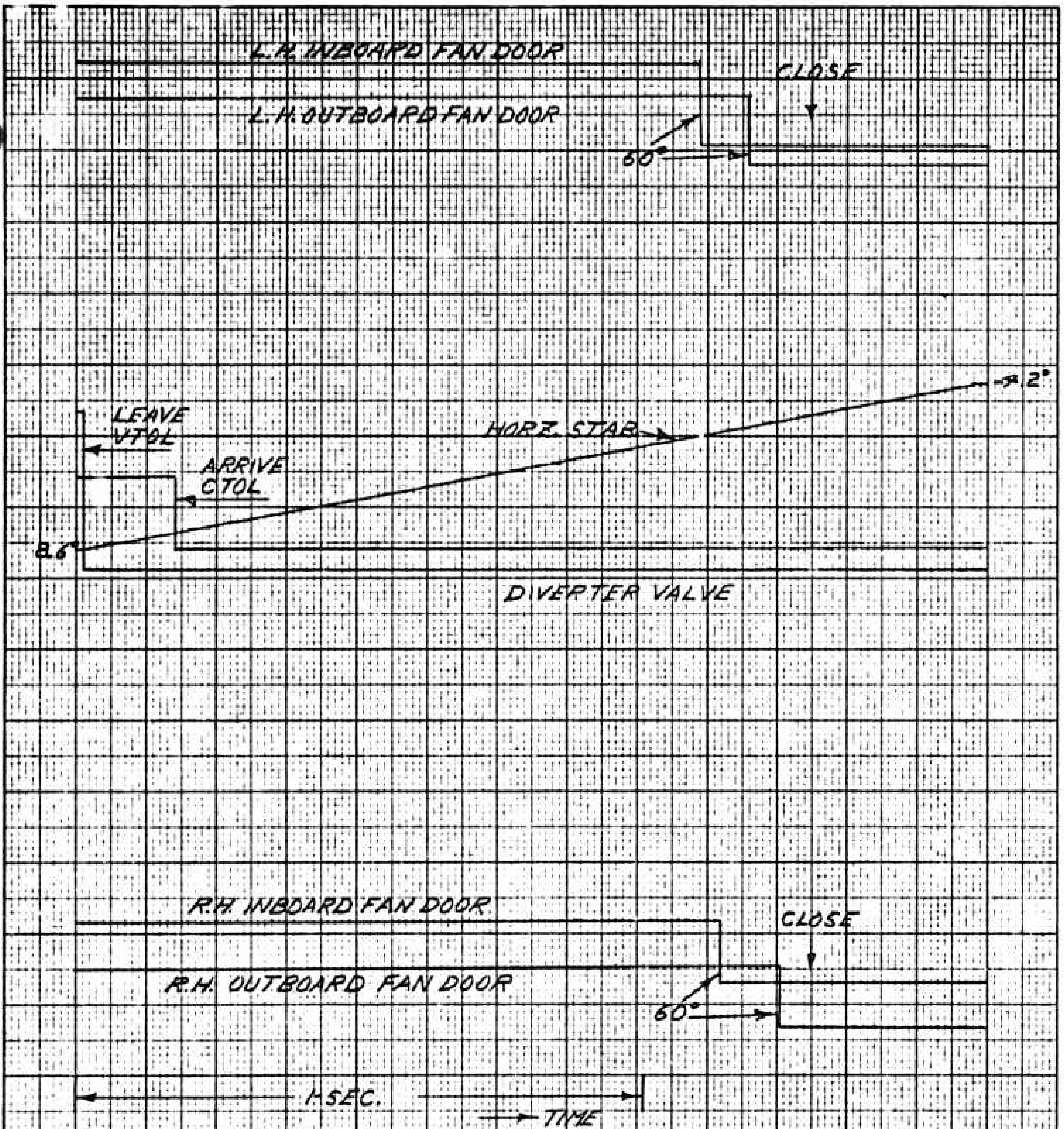
NO. 1 A/C
FLIGHT MODE CONVERSION
CTOL TO VTOL
GROUND POWER, NO. 2 HYD. SYSTEM ONLY
6 MARCH 64

FIGURE - 6.3.107



NO. 1 A/C
FLIGHT MODE CONVERSION
ABORT TEST, VTOL TO CTOL TO VTOL
GROUND POWER, BOTH HYD. SYSTEMS
6 MARCH 64

FIGURE-6.3.108



NO. 1 A/C
FLIGHT MODE CONVERSION
VTOL TO CTOL
BOTH ENGINES 70% RPM
7 MARCH 64

FIGURE-6.3.109

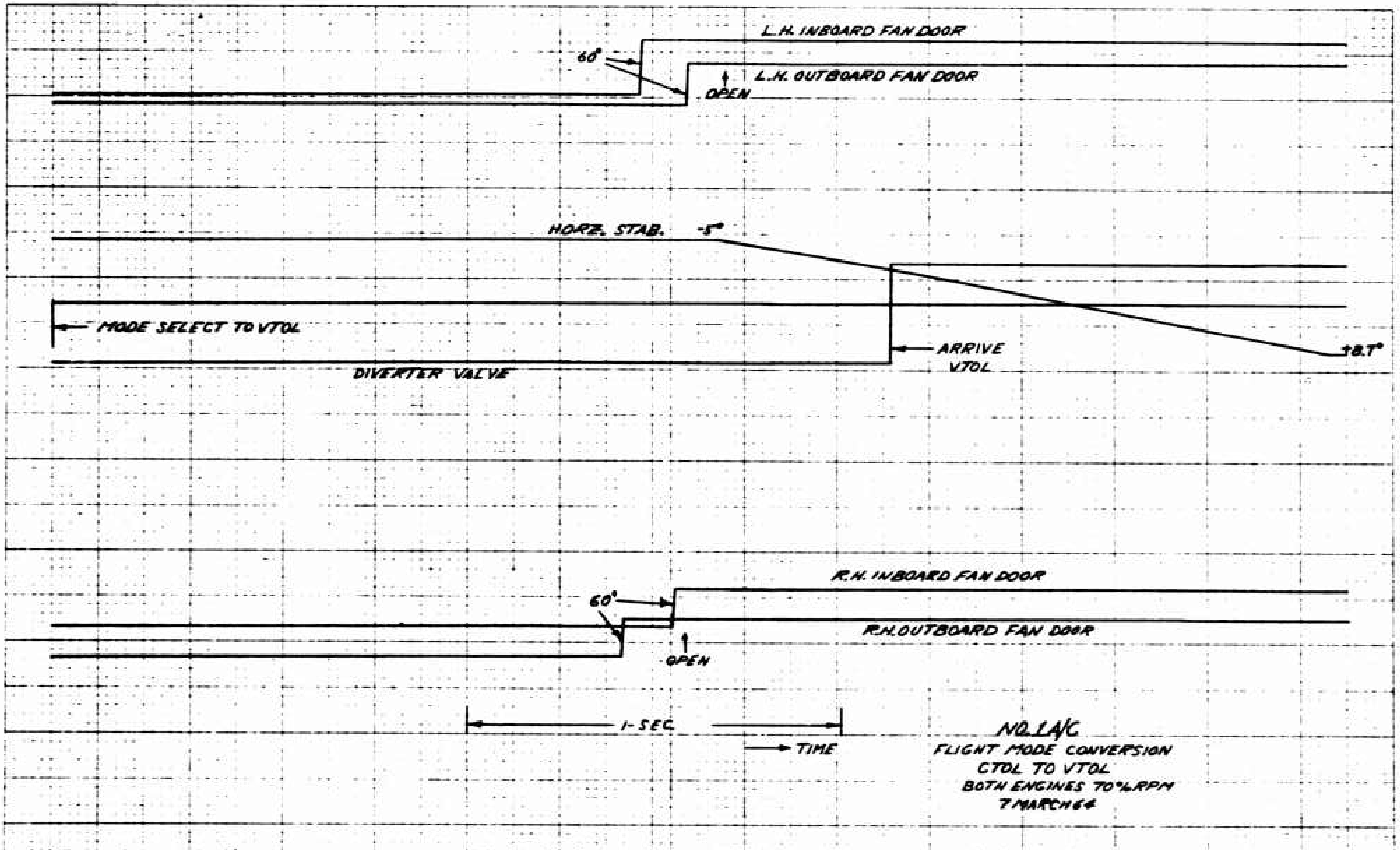


FIGURE- 6.3.110

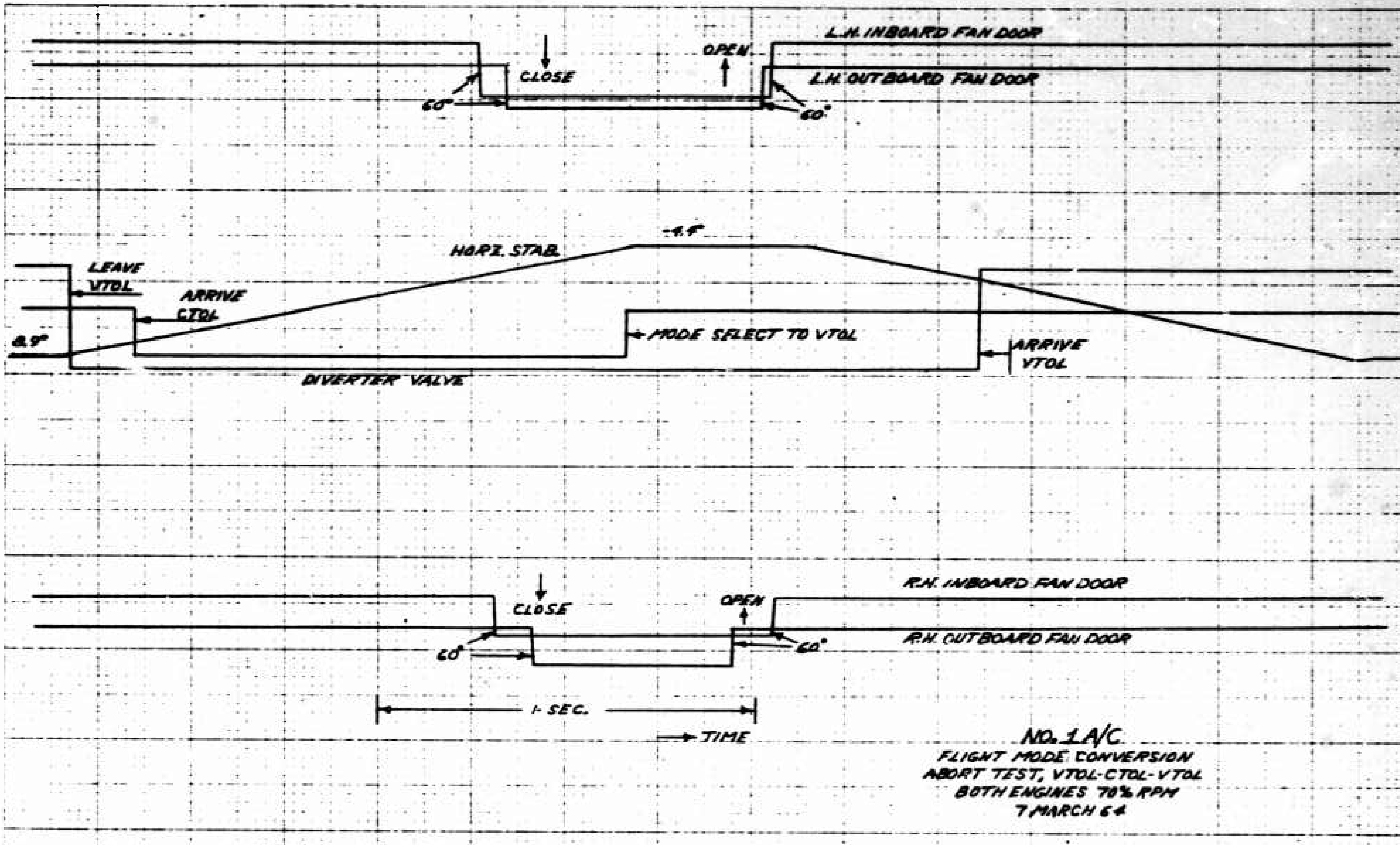
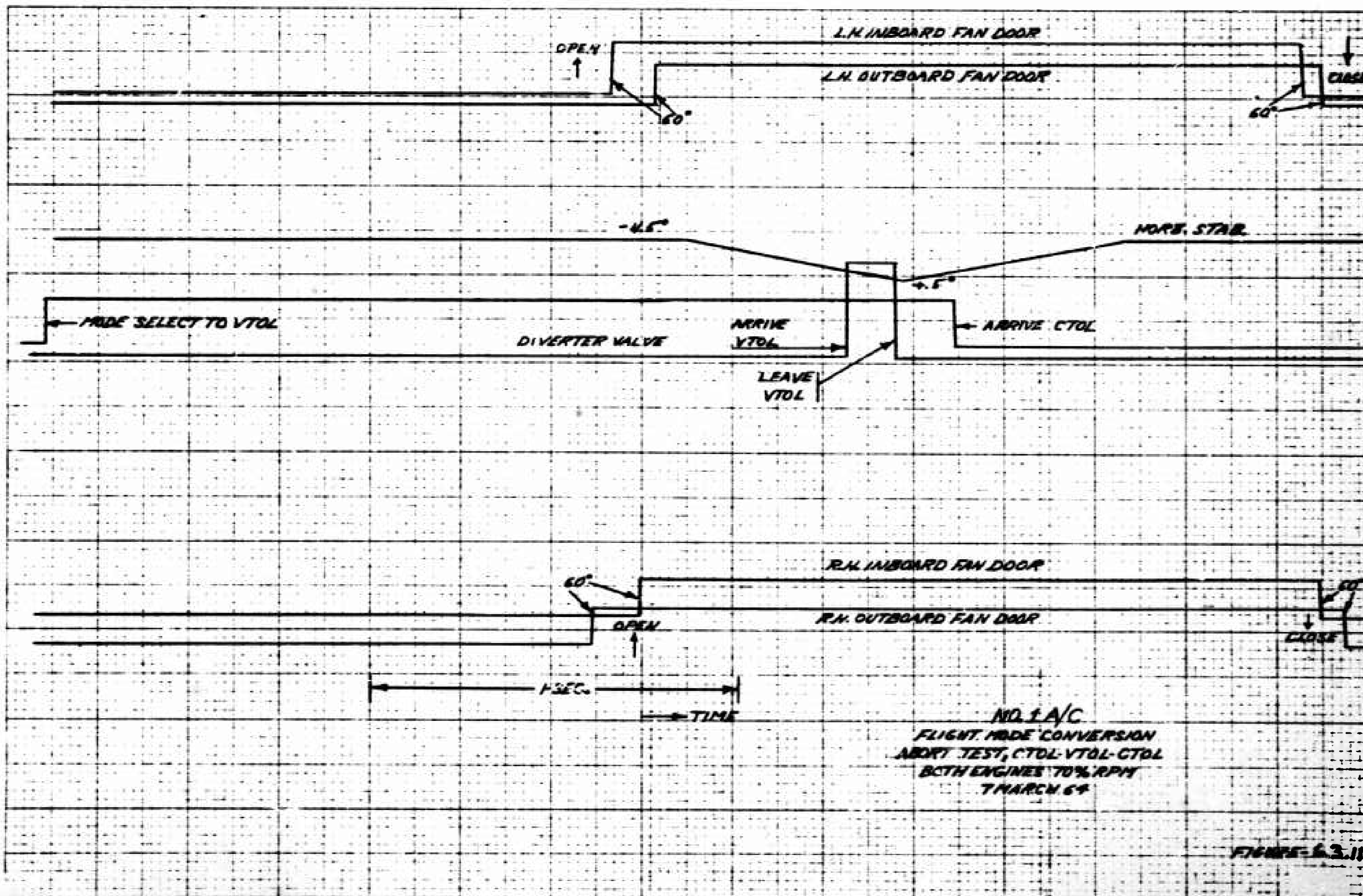
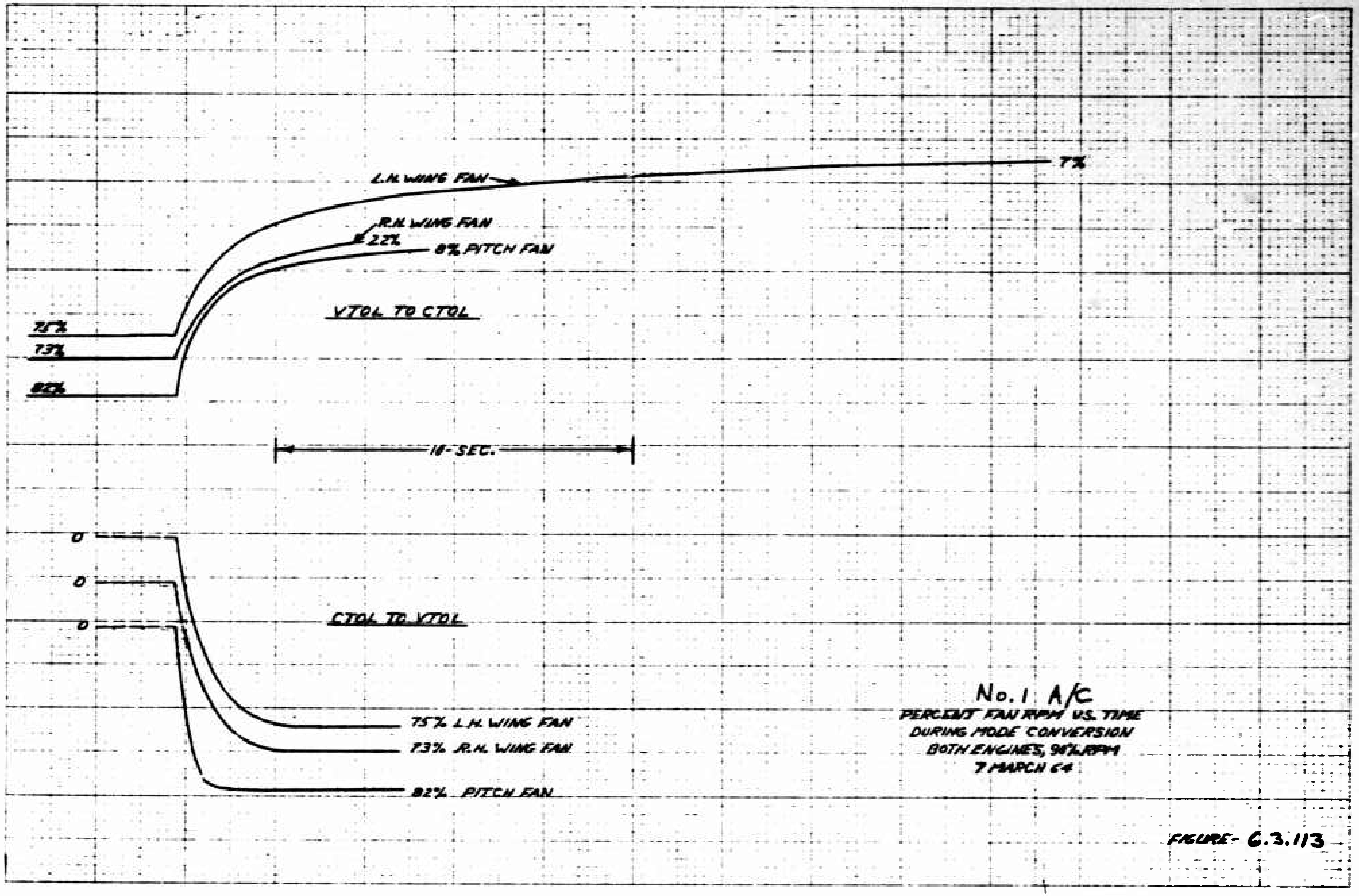


FIGURE- 6.3.111



NO. 1 A/C
FLIGHT MODE CONVERSION
ABORT TEST, CTOL-VTOL-CTOL
BOTH ENGINES 70% RPM
7 MARCH 64

FIGURE 6.3.112



No. 1 A/C
PERCENT FAN RPM VS. TIME
DURING MODE CONVERSION
BOTH ENGINES, 96% RPM
7 MARCH 64

FIGURE- 6.3.113

NO. 1 A/C

PERCENT FAN RPM VS. TIME

ABORT TEST, VTOL TO CTOL TO VTOL

BOTH ENGINES, 90% RPM

7 MARCH 64

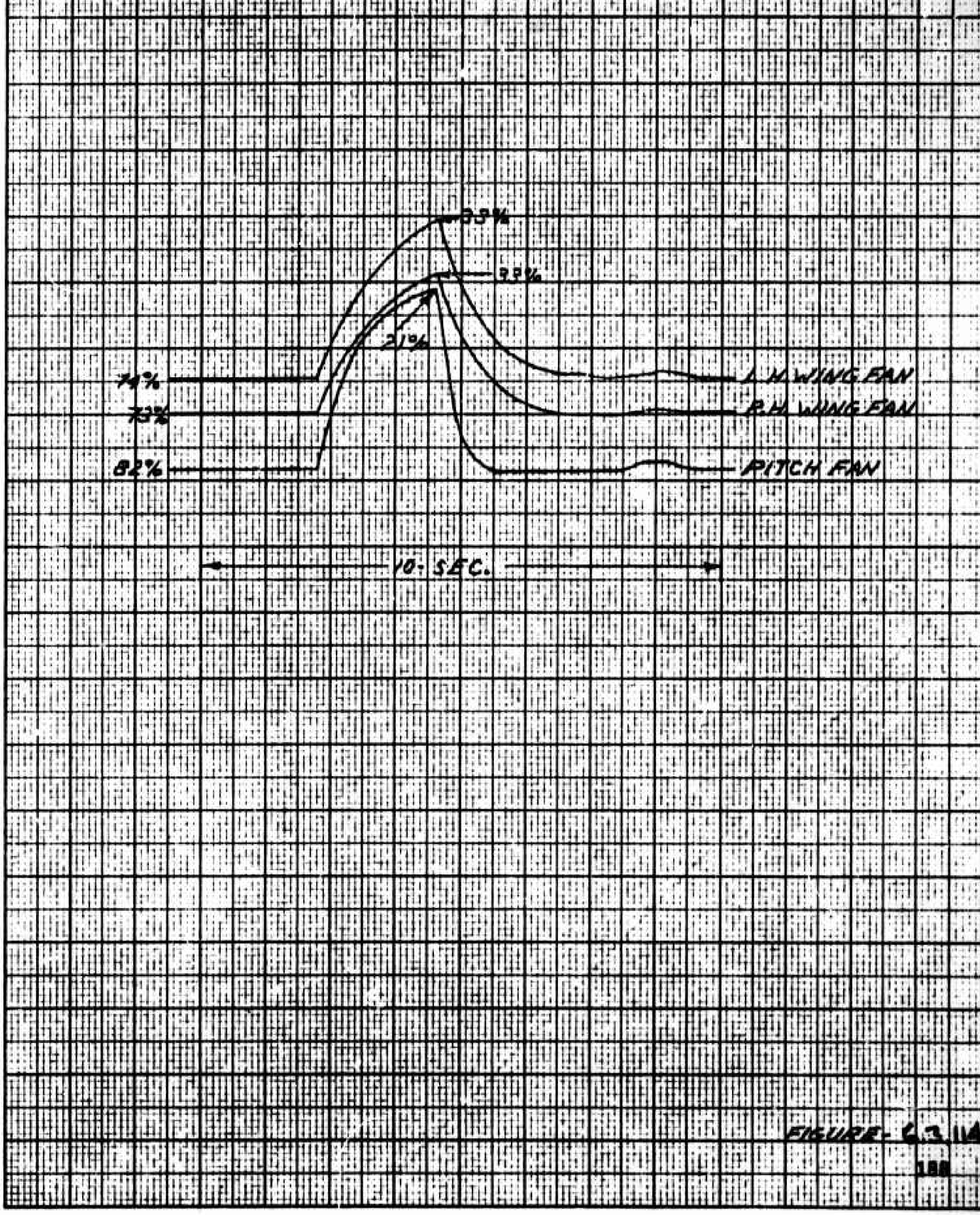


FIGURE- 6.3.114

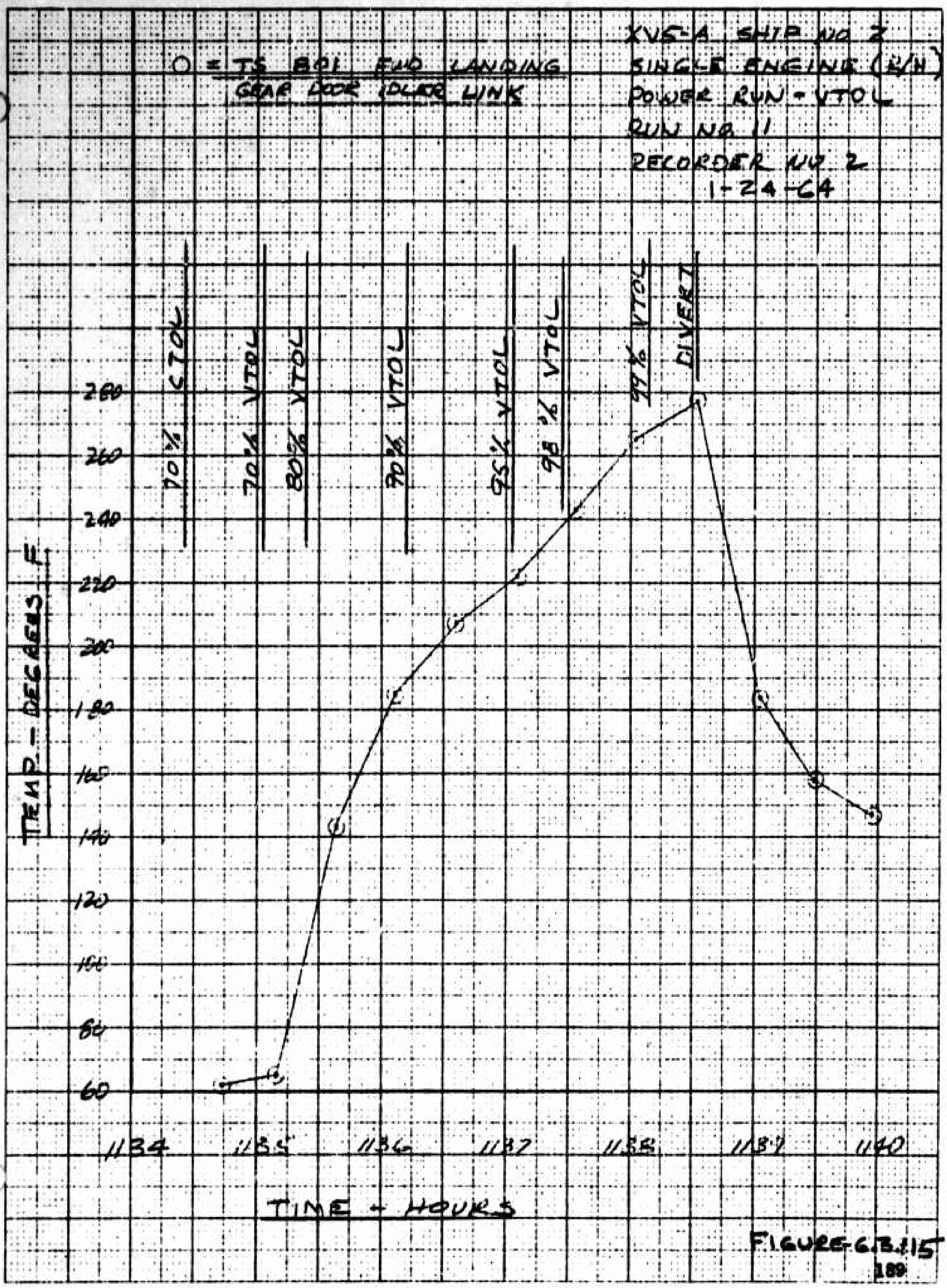
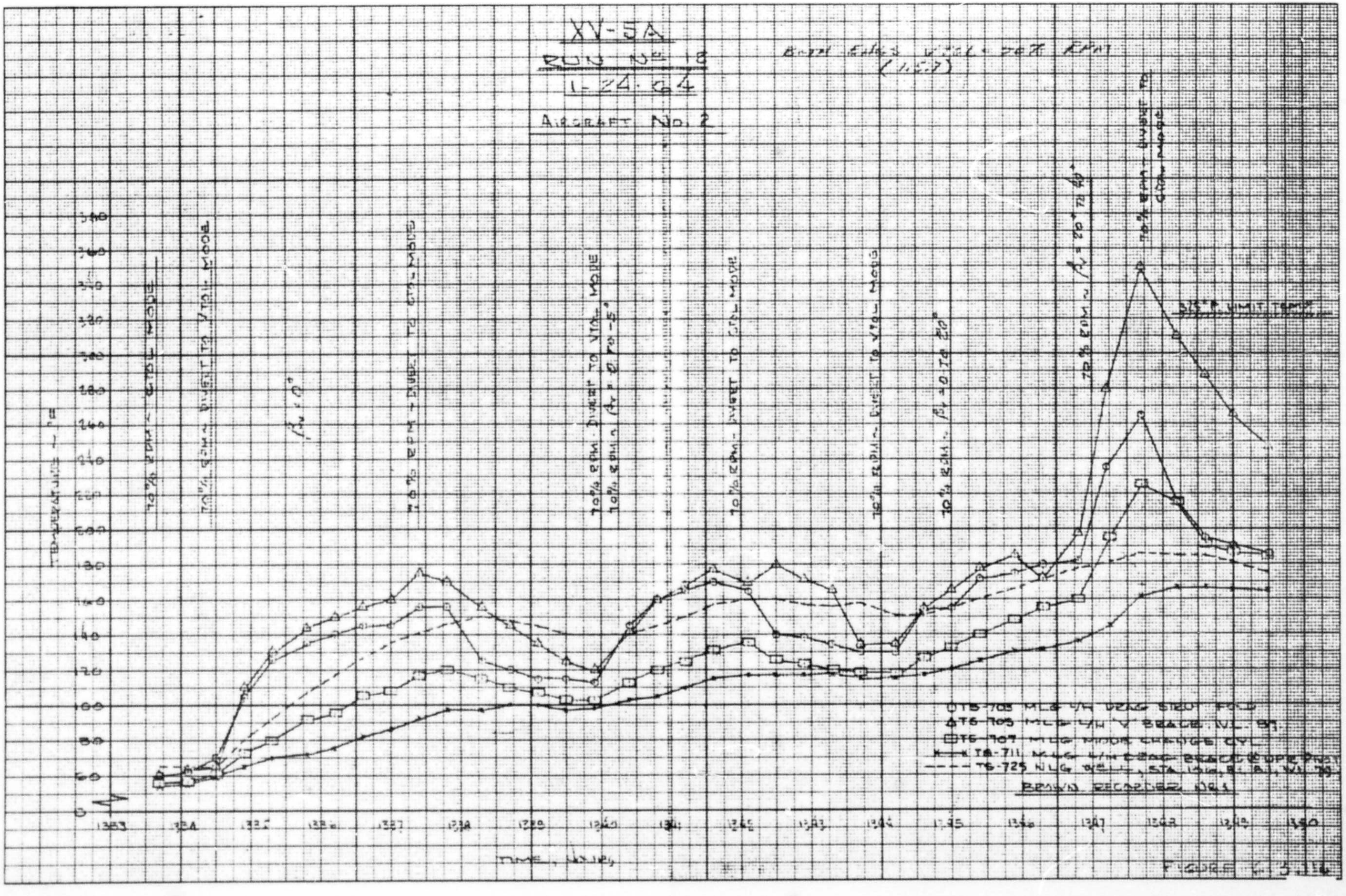
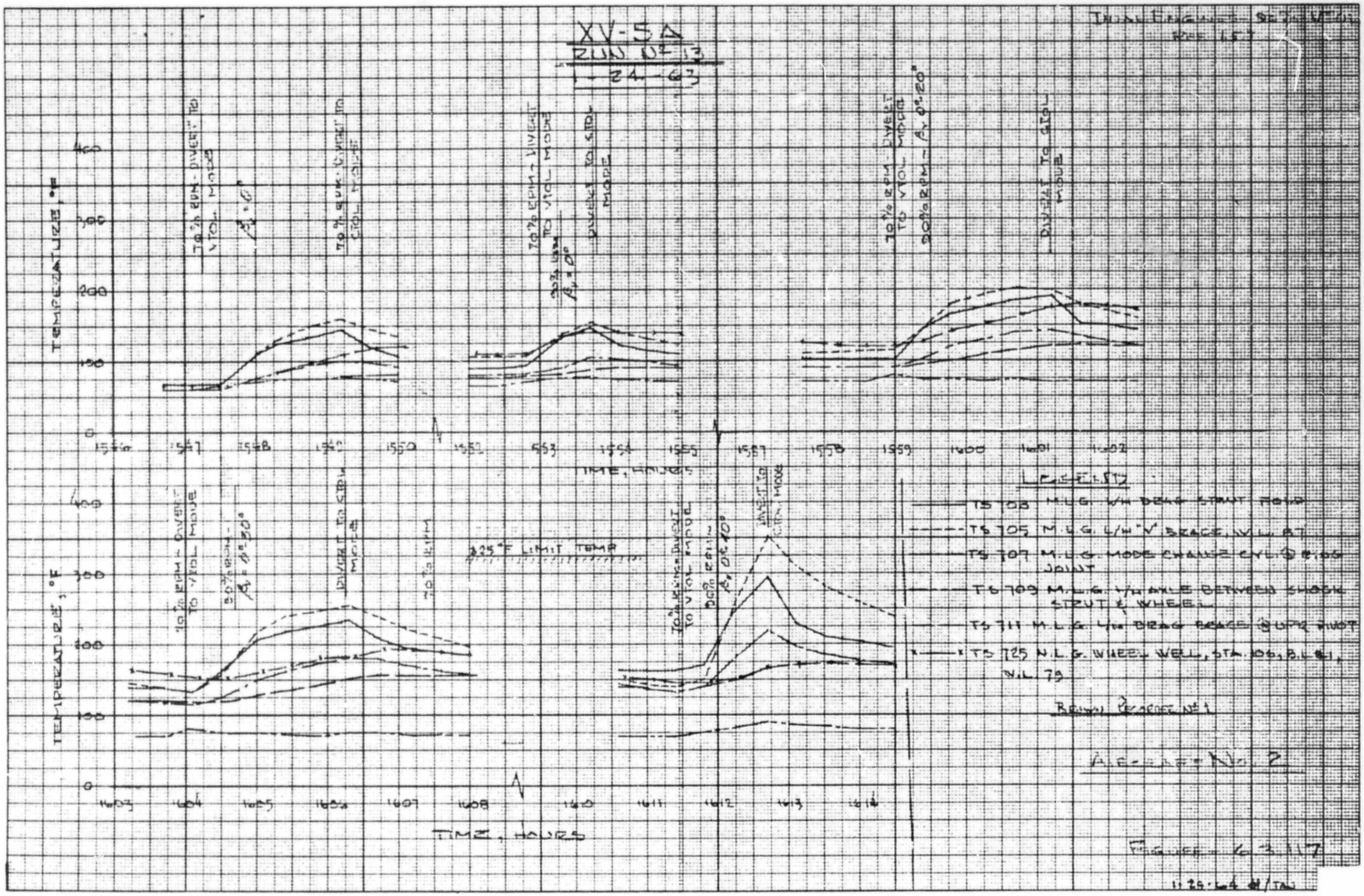

○ = TS 801 FWD LANDING GEAR DOOR IDLER LINK
XV5-A SHIP NO 2
SINGLE ENGINE (L/H)
POWER RUN - VTOL
RUN NO. 11
RECORDER NO 2
1-24-64
70% CTOL
70% VTOL
80% VTOL
90% VTOL
95% VTOL
98% VTOL
99% VTOL
DIVERT
TEMP - DEGREES F
TIME - HOURS
60
80
100
120
140
160
180
200
220
240
260
280
1134
1135
1136
1137
1138
1139
1140

FIGURE 6.B.15


XV-5A
RUN No 18
1-24-64
AIRCRAFT No. 2
TEMPERATURE ~ °F
TIME, HOURS
BROWN RECORDER No. 1


XV-5A
TEMPERATURE, °F
TIME, HOURS
225°F LIMIT TEMP
LEGEND
TS 705 M.L.G. L/H "V" BRACE
TS 711 M.L.G. L/H DRAG BRACE @ UPR PIVOT
TS 725 N.L.G. WHEEL WELL
AIRCRAFT No. 2
FIGURE 6.3.117

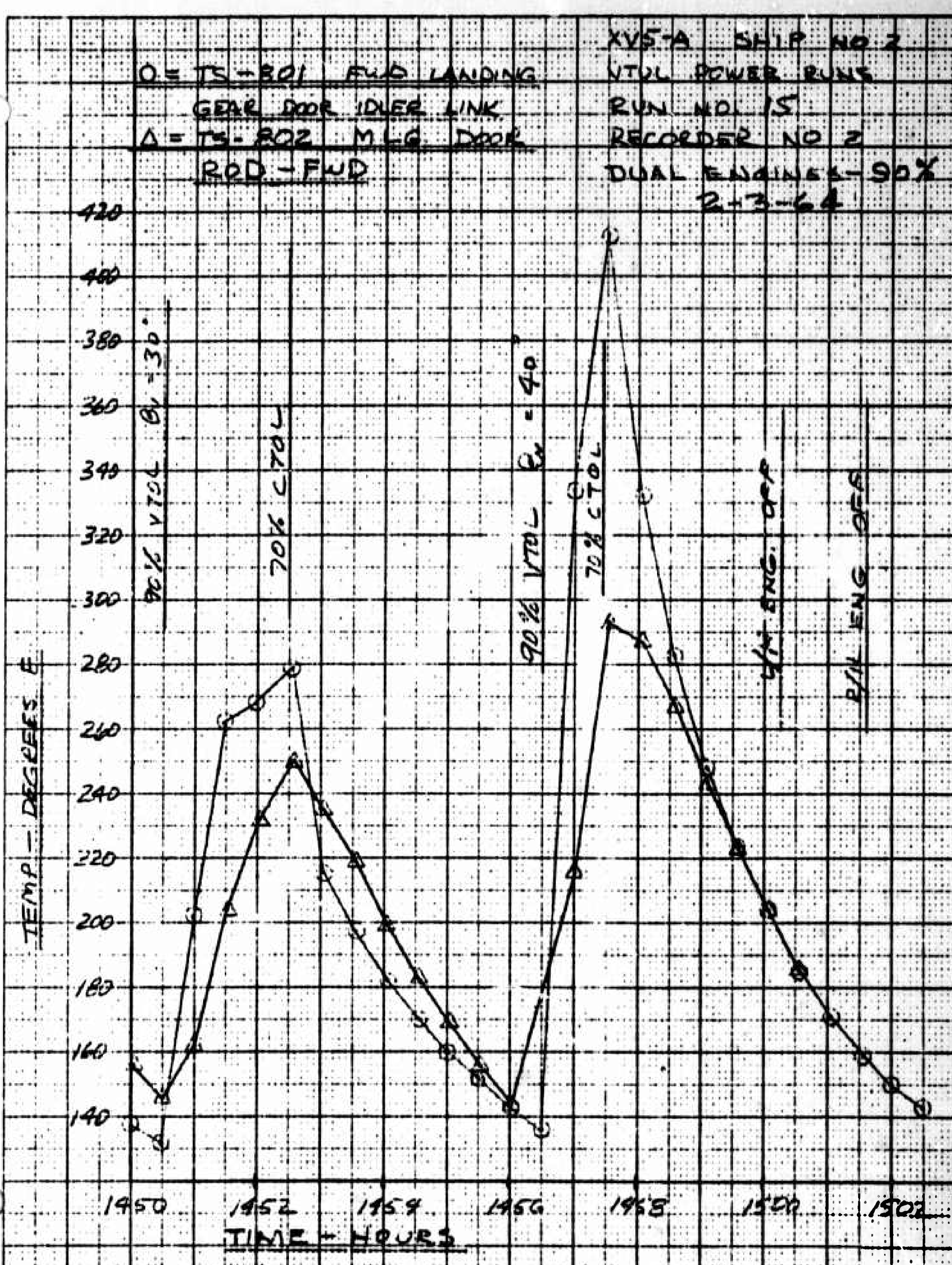


FIGURE - 6.3.118

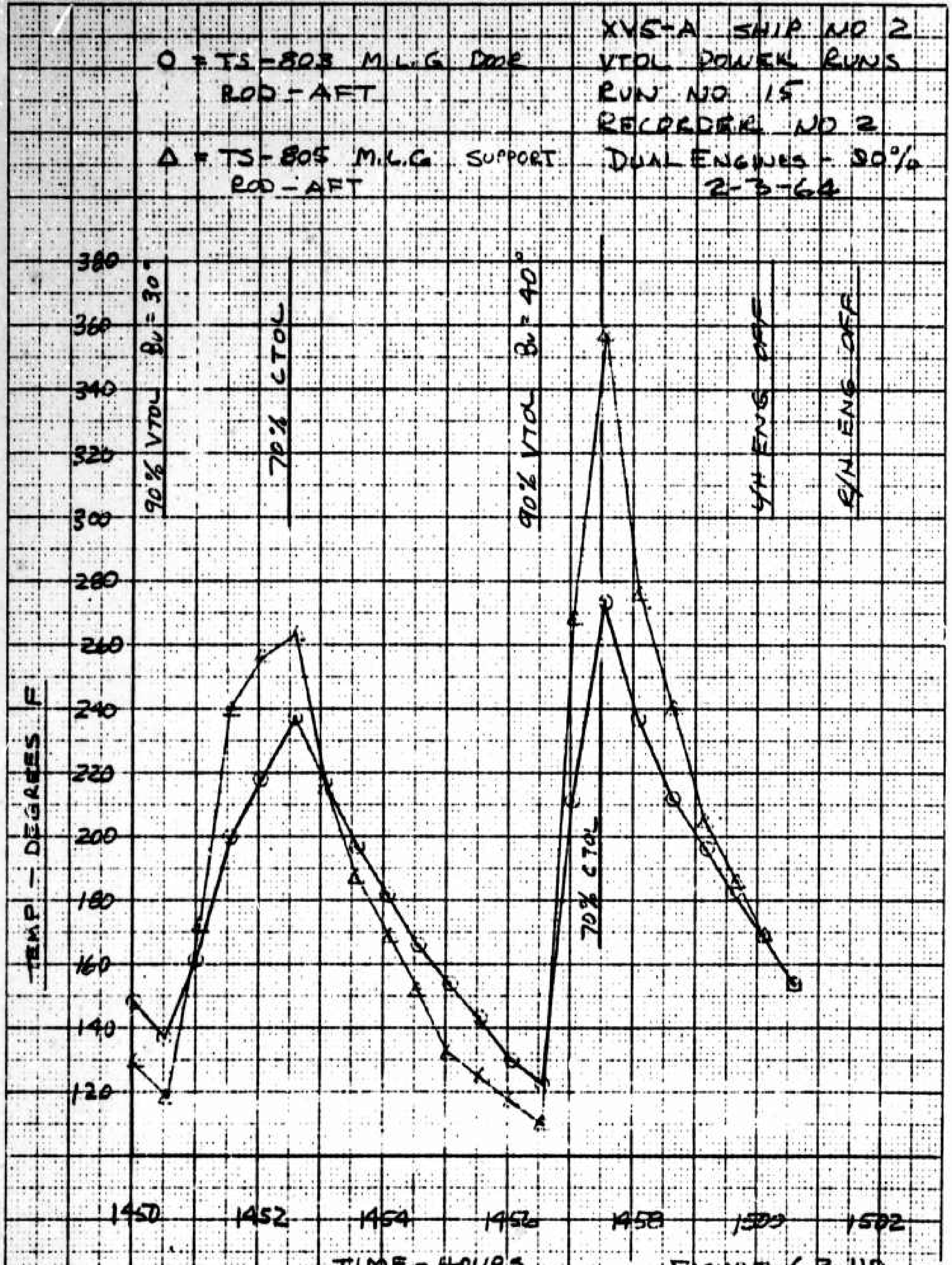


FIGURE 6.3.119

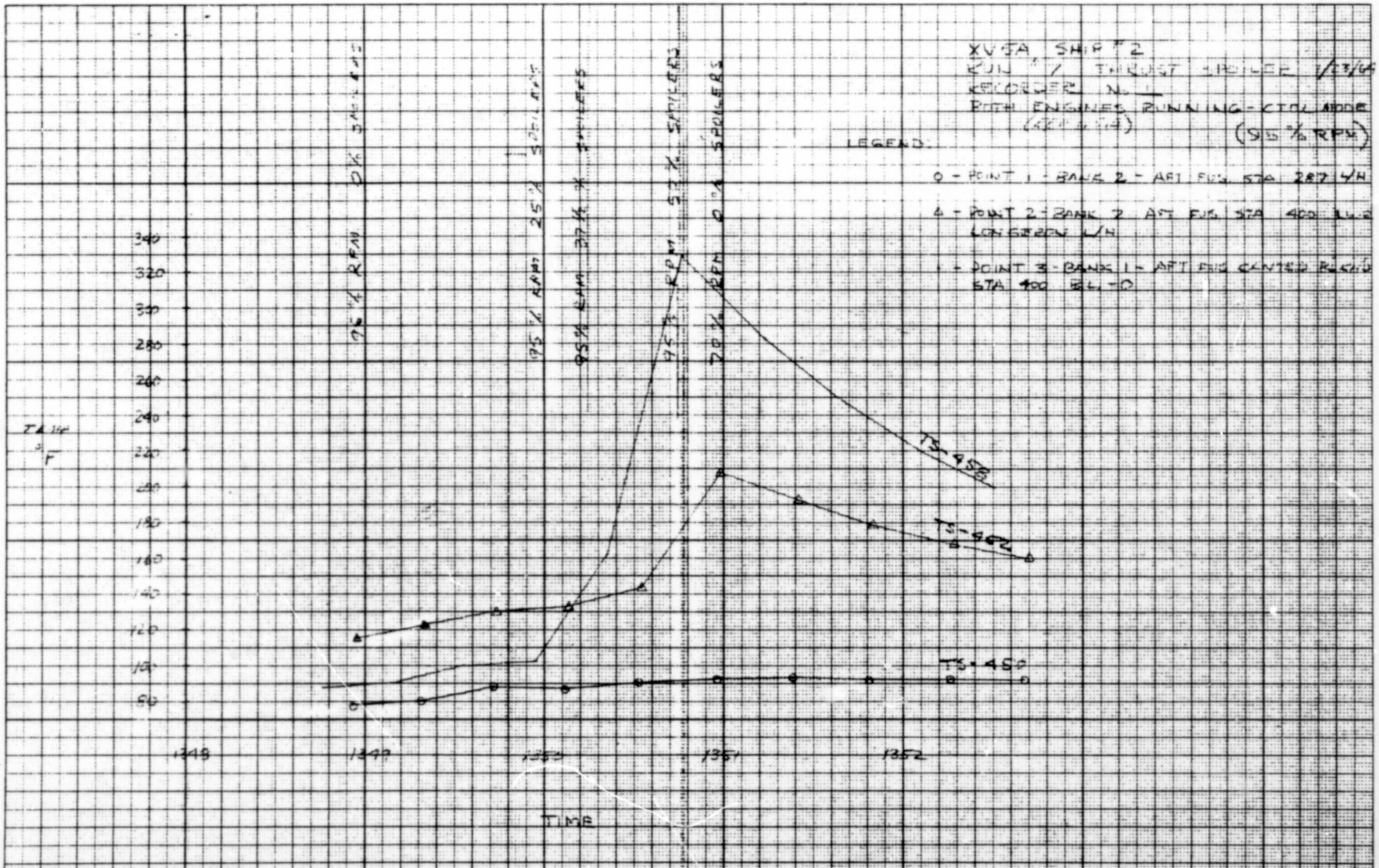

XV-5A, SHIP #2
RUN 7 THRUST SPOILER 1/23/64
RECORDER No. 1
BOTH ENGINES RUNNING - CTOL MODE
(95% RPM)
LEGEND:
o - POINT 1 - BANK 2 - AFT FUS. STA 287 L/H
Δ - POINT 2 - BANK 2 AFT FUS. STA 400
LONGERON L/H
+ - POINT 3 - BANK 1 - AFT FUS CANTED
STA 400 BL -0
95% RPM 0% SPOILERS
95% RPM 25% SPOILERS
95% RPM 37% SPOILERS
45% RPM 57% SPOILERS
70% RPM 0% SPOILERS
TS-458
TS-462
TS-450
340
320
300
280
260
240
220
200
180
160
140
120
100
80
°F
1348
1349
1350
1351
1352
TIME

FIGURE-6.3.20

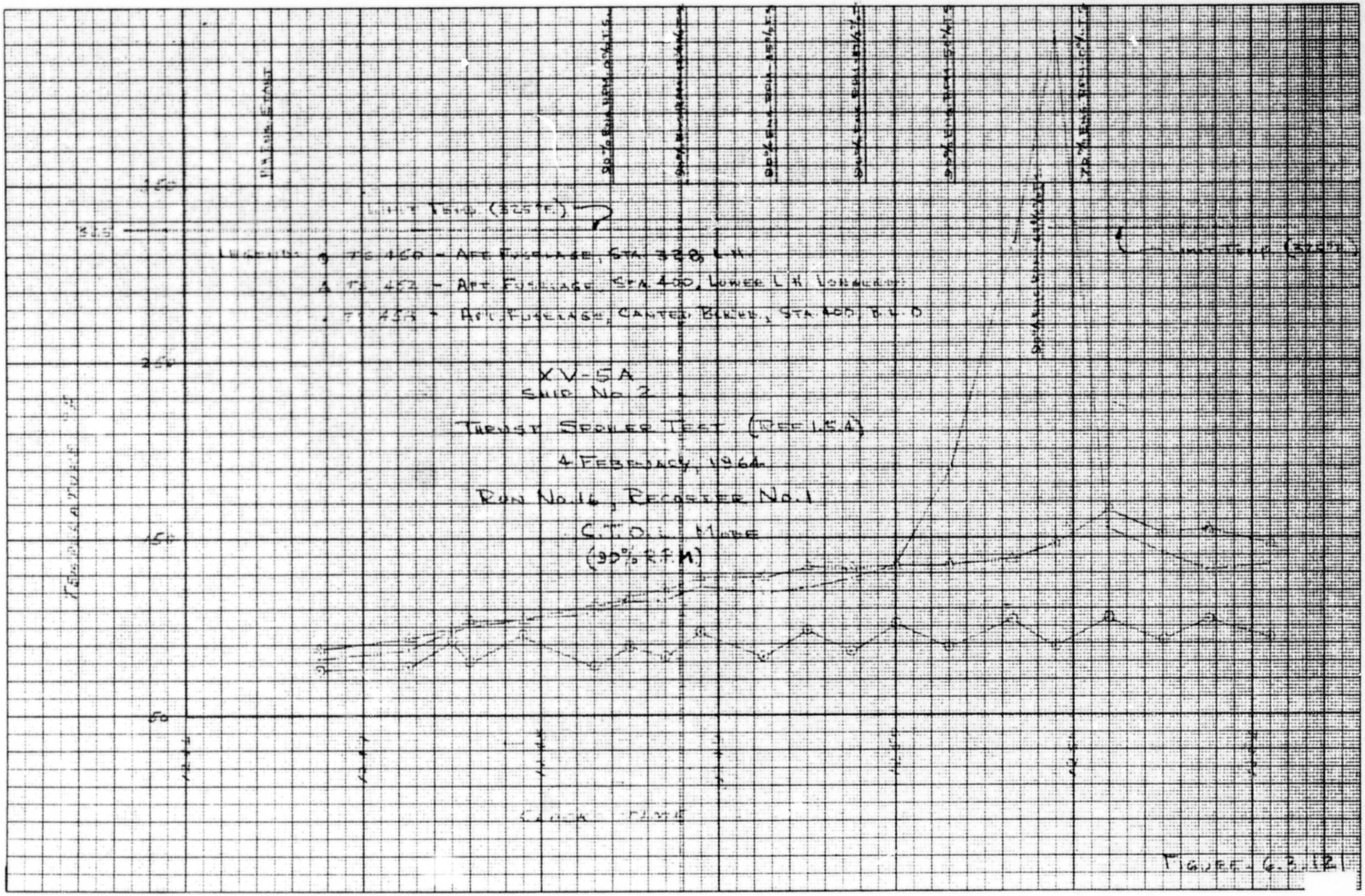


FIGURE 6.3.12

T.L.V./J.K.G.

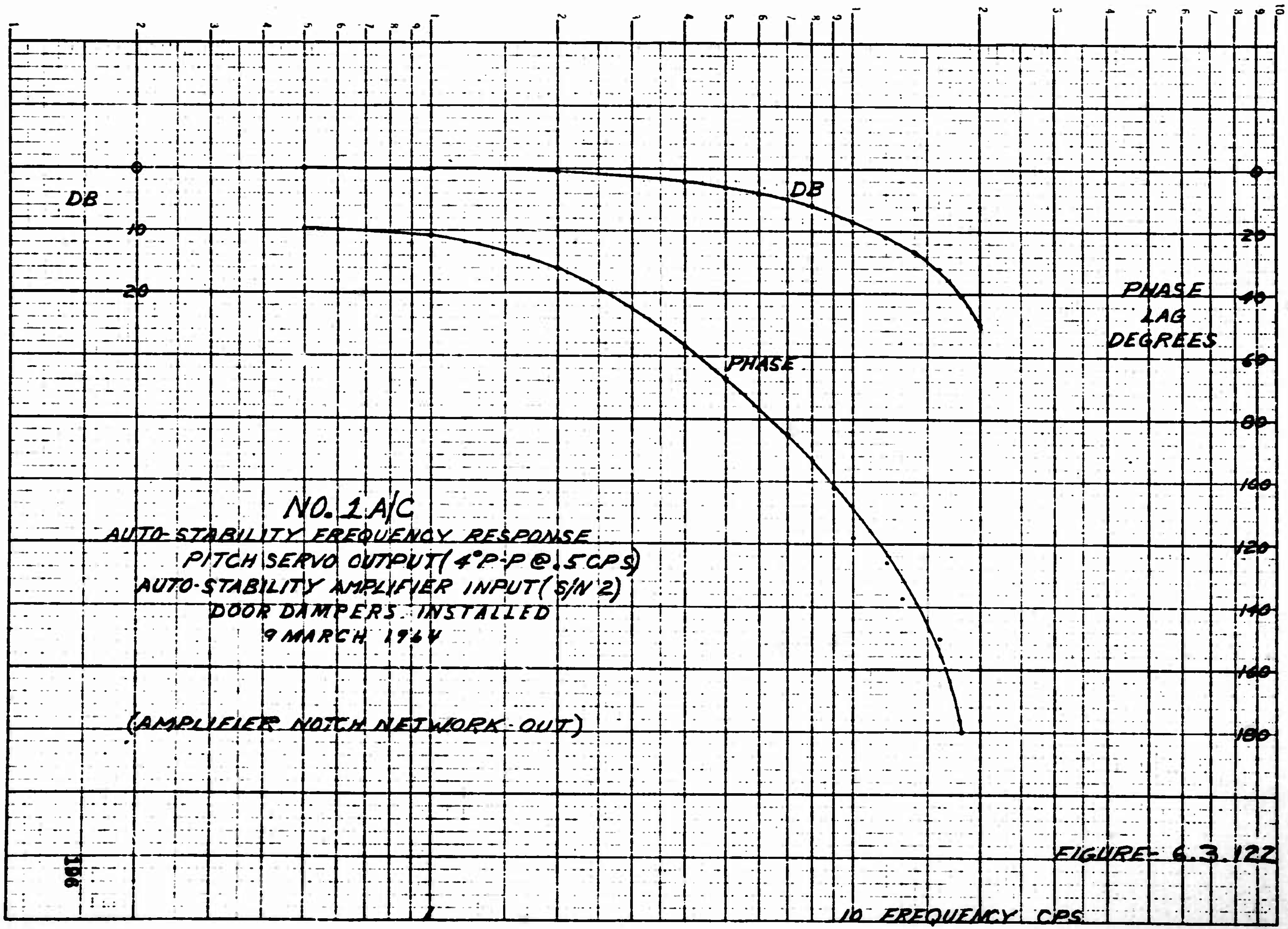


FIGURE 6.3.122

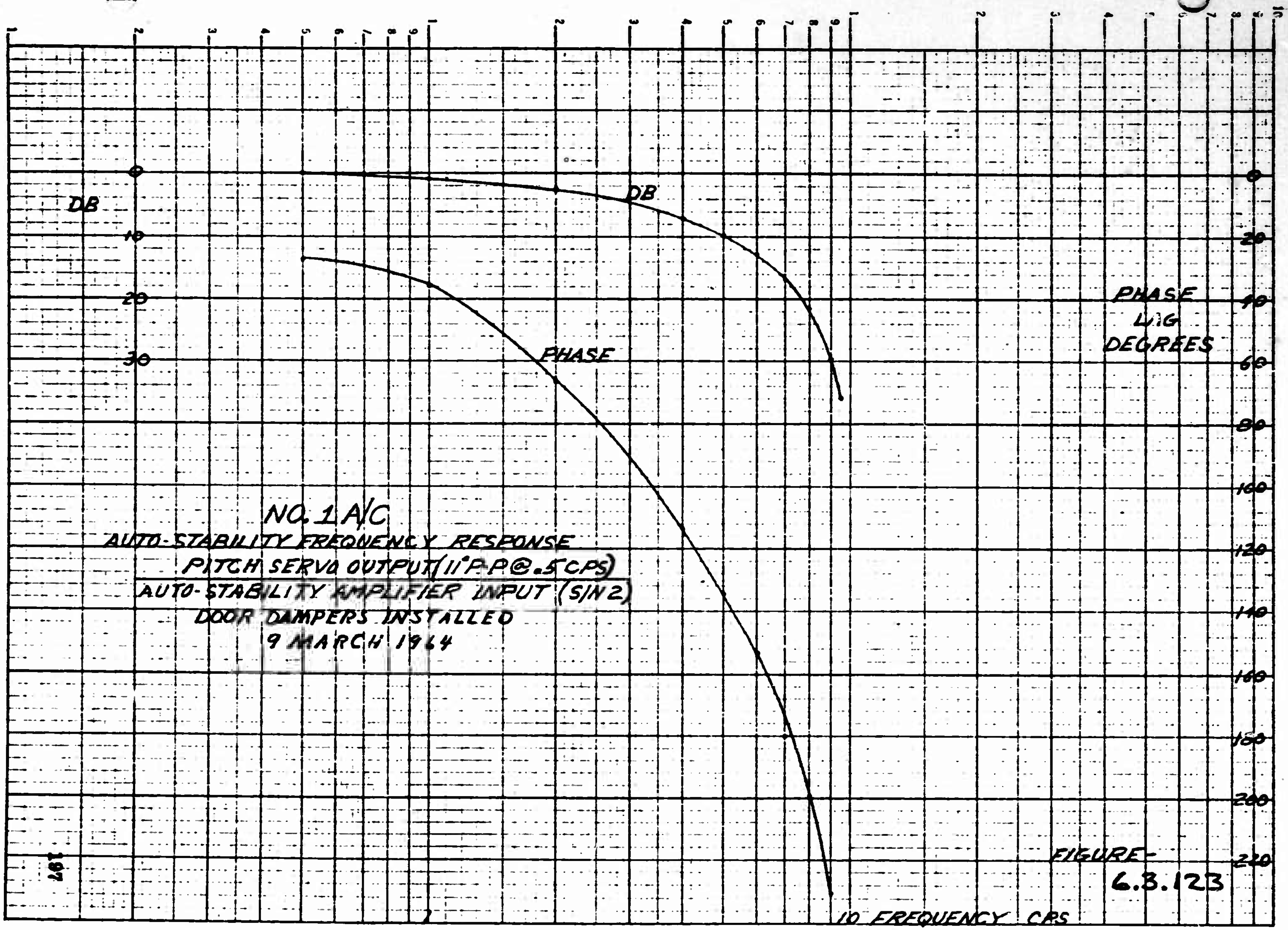


FIGURE- 6.3.123

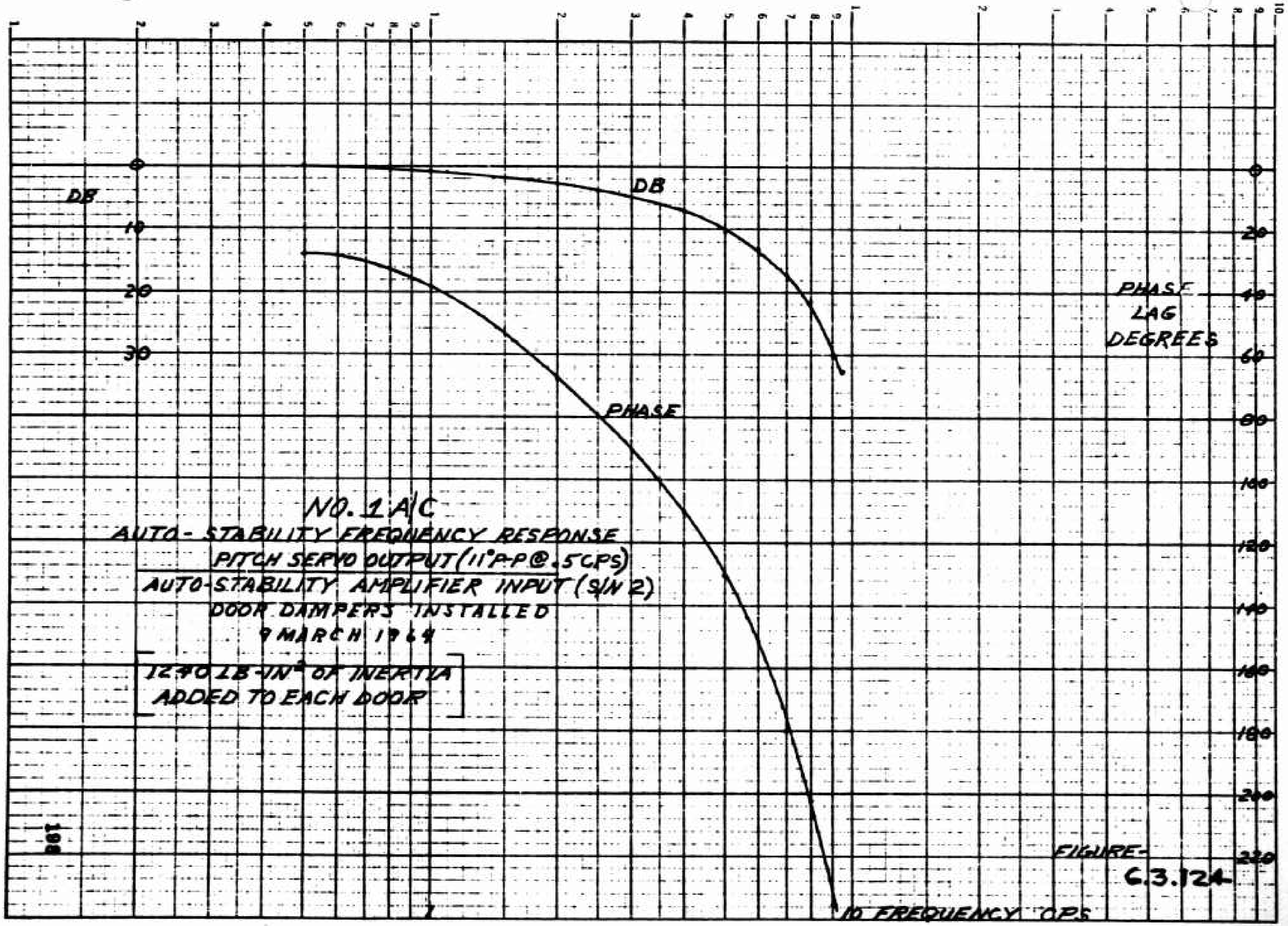
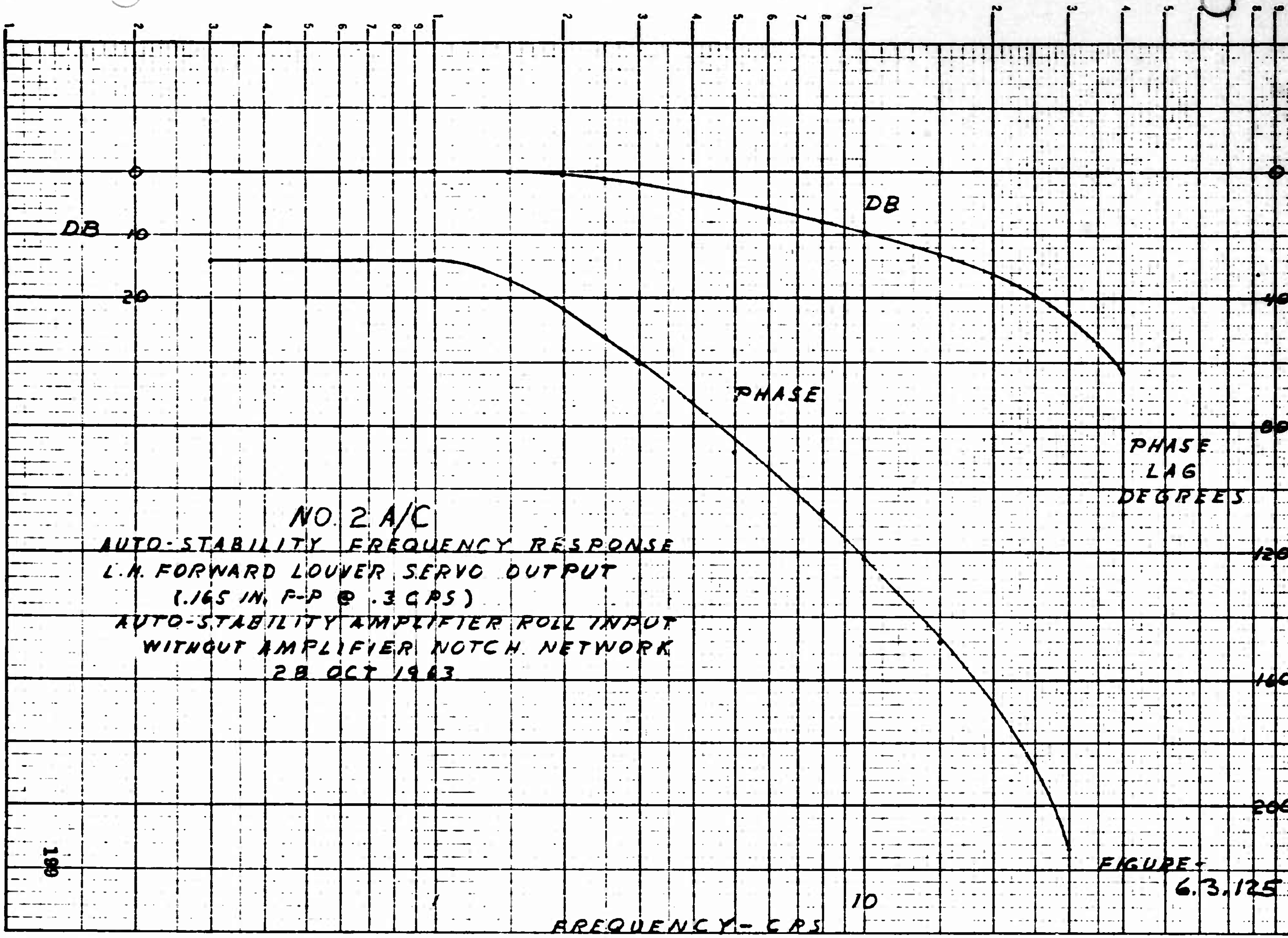


FIGURE- C.3.124


NO. 2 A/C
AUTO-STABILITY FREQUENCY RESPONSE
L.H. FORWARD LOUVER SERVO OUTPUT
(.165 IN. F-P @ .3 CPS)
AUTO-STABILITY AMPLIFIER ROLL INPUT
WITHOUT AMPLIFIER NOTCH NETWORK
28 OCT 1963
DB
PHASE
PHASE LAG DEGREES
0
40
80
120
160
200
DB
10
20
1
10
FREQUENCY - CPS
FIGURE - 6.3.125

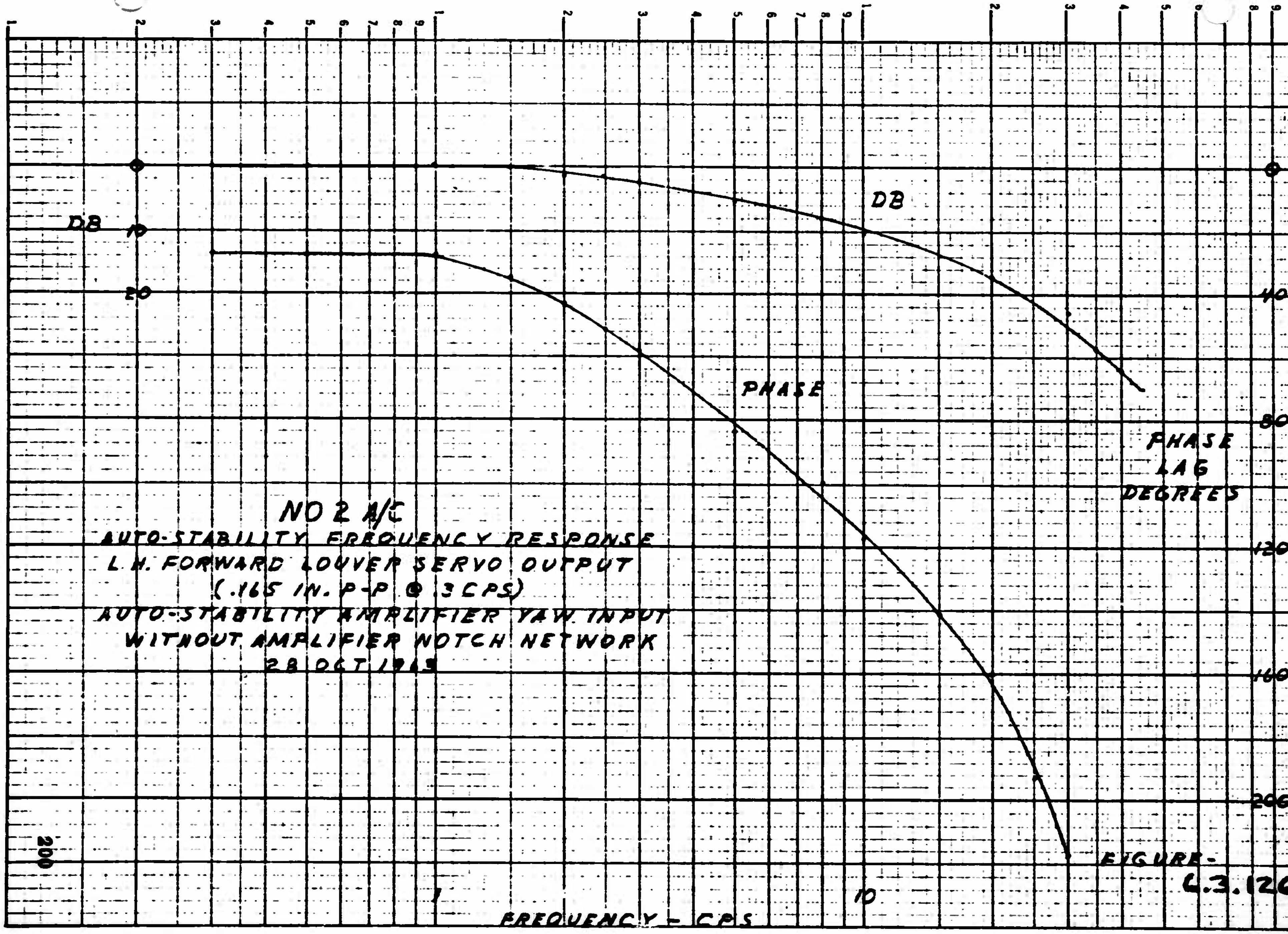


FIGURE- 6.3.126

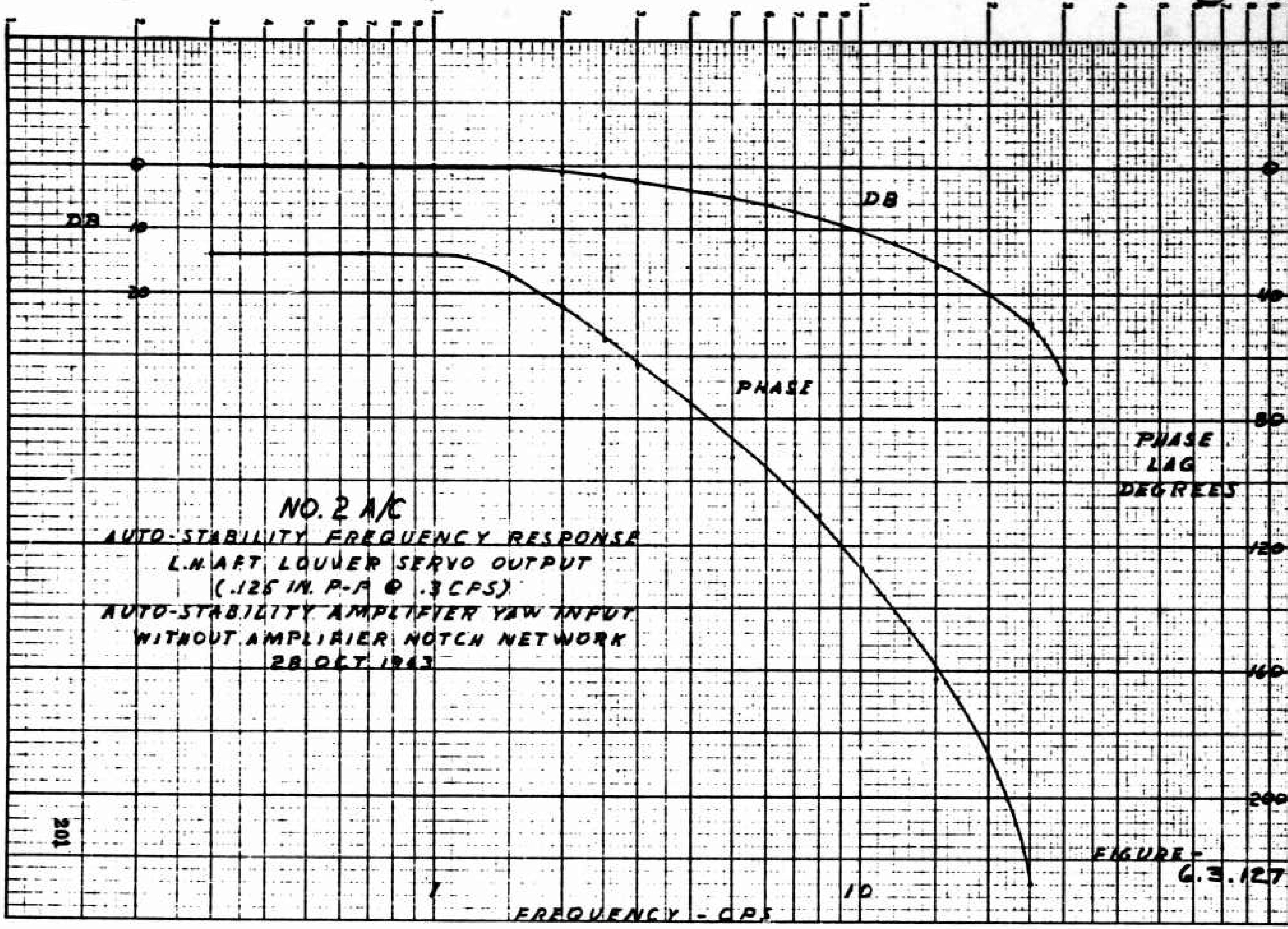


FIGURE - 6.3.127

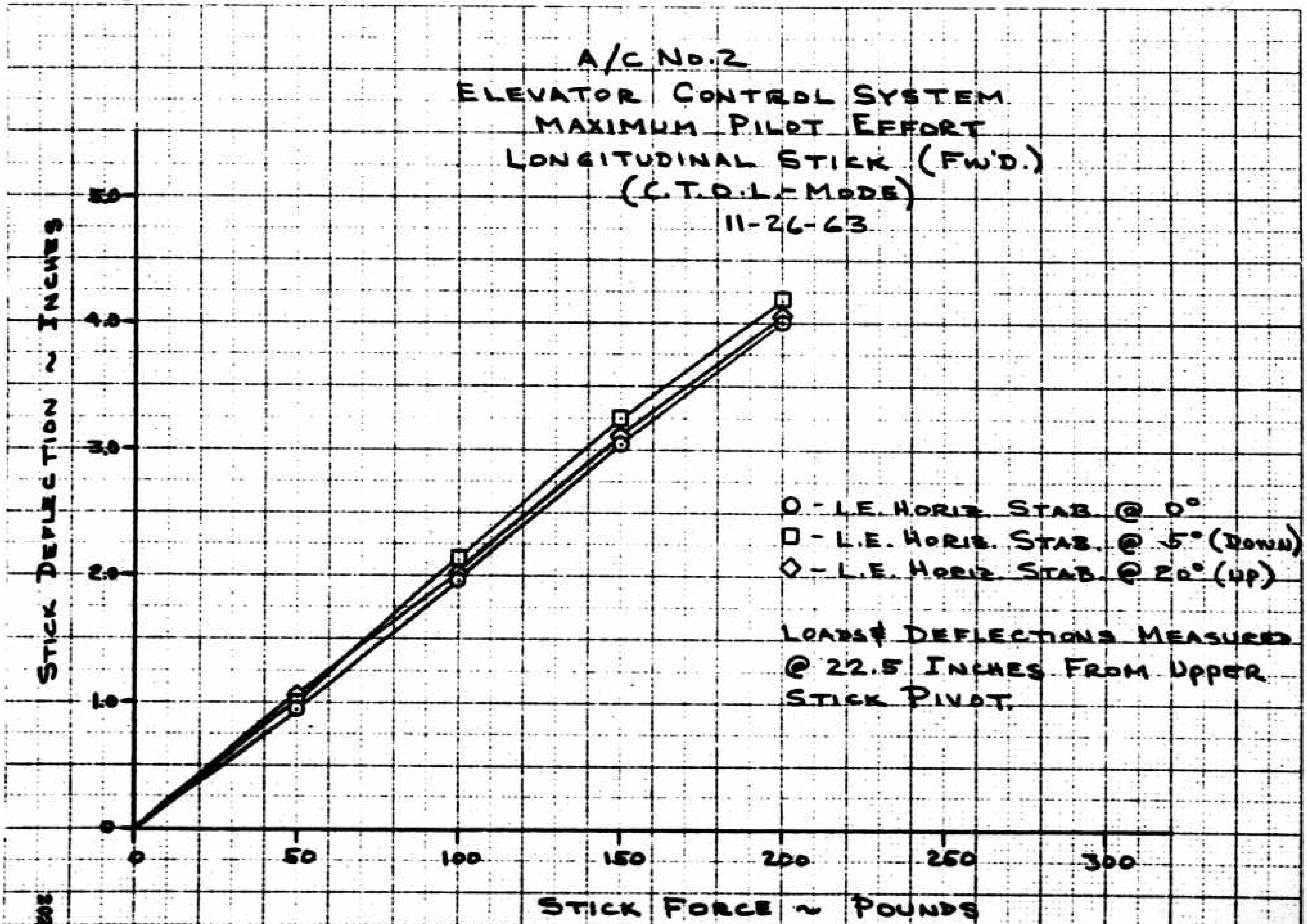


Figure-6.3.128

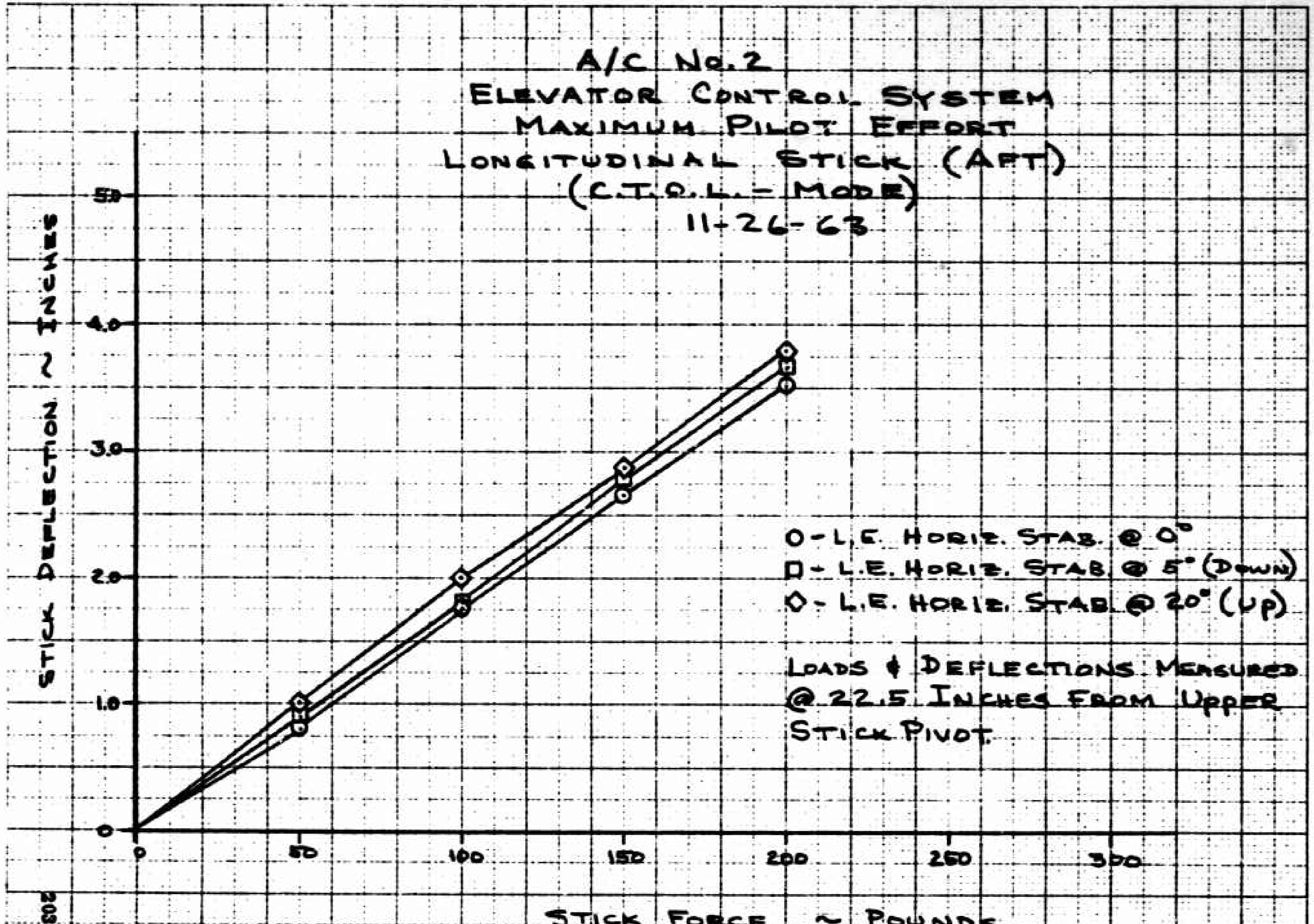


FIGURE- 6.3.123

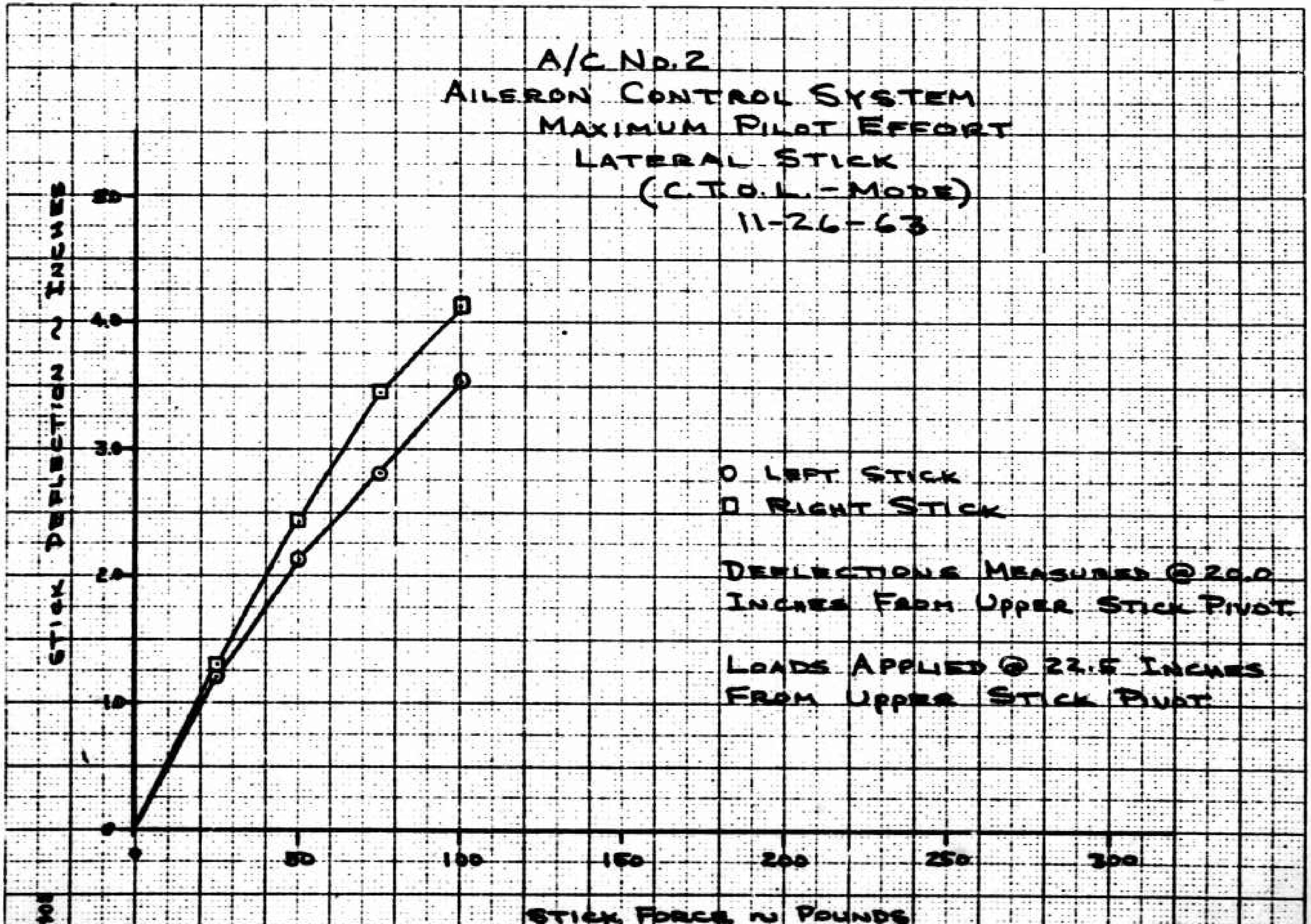


FIGURE-6.3.130

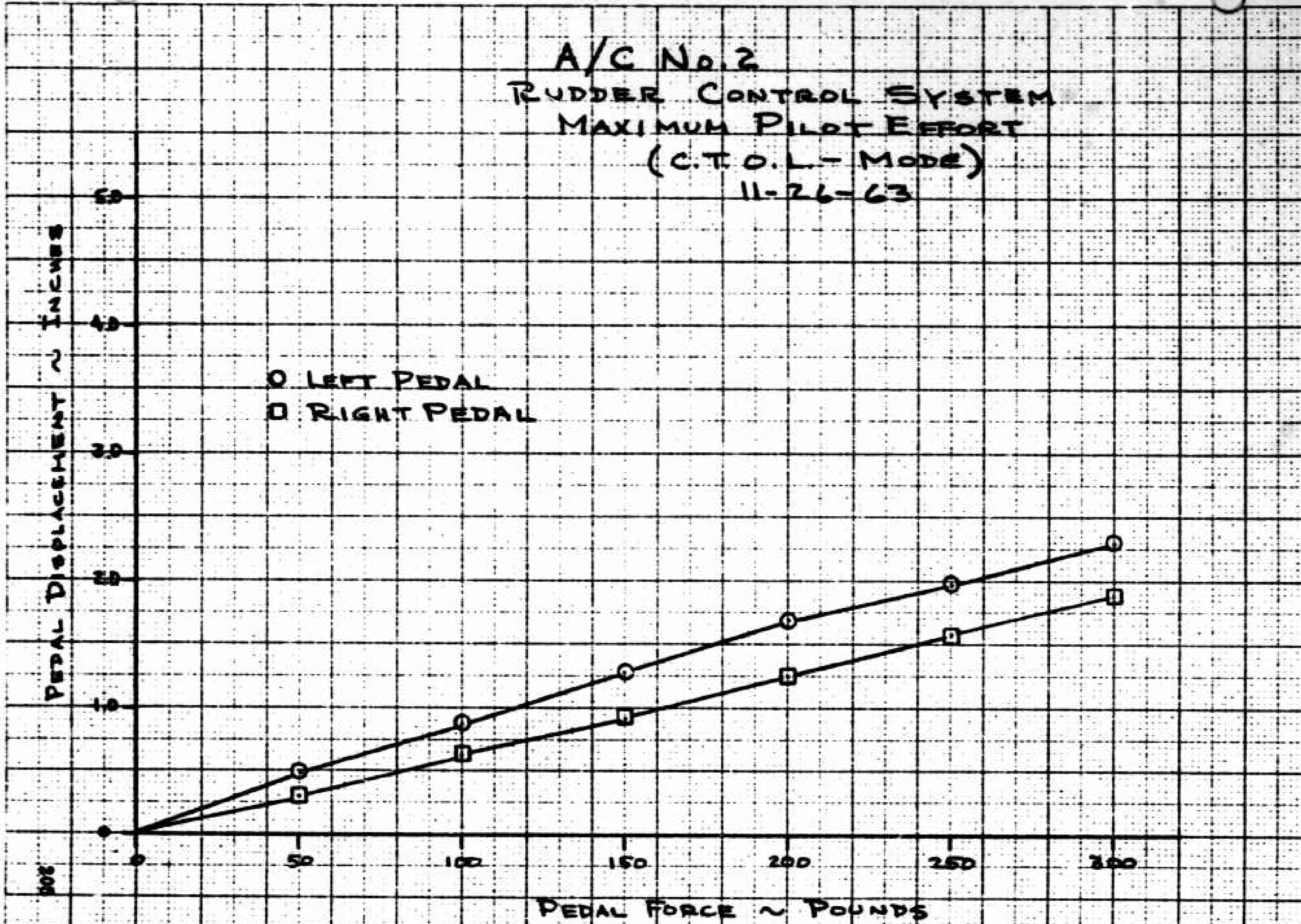
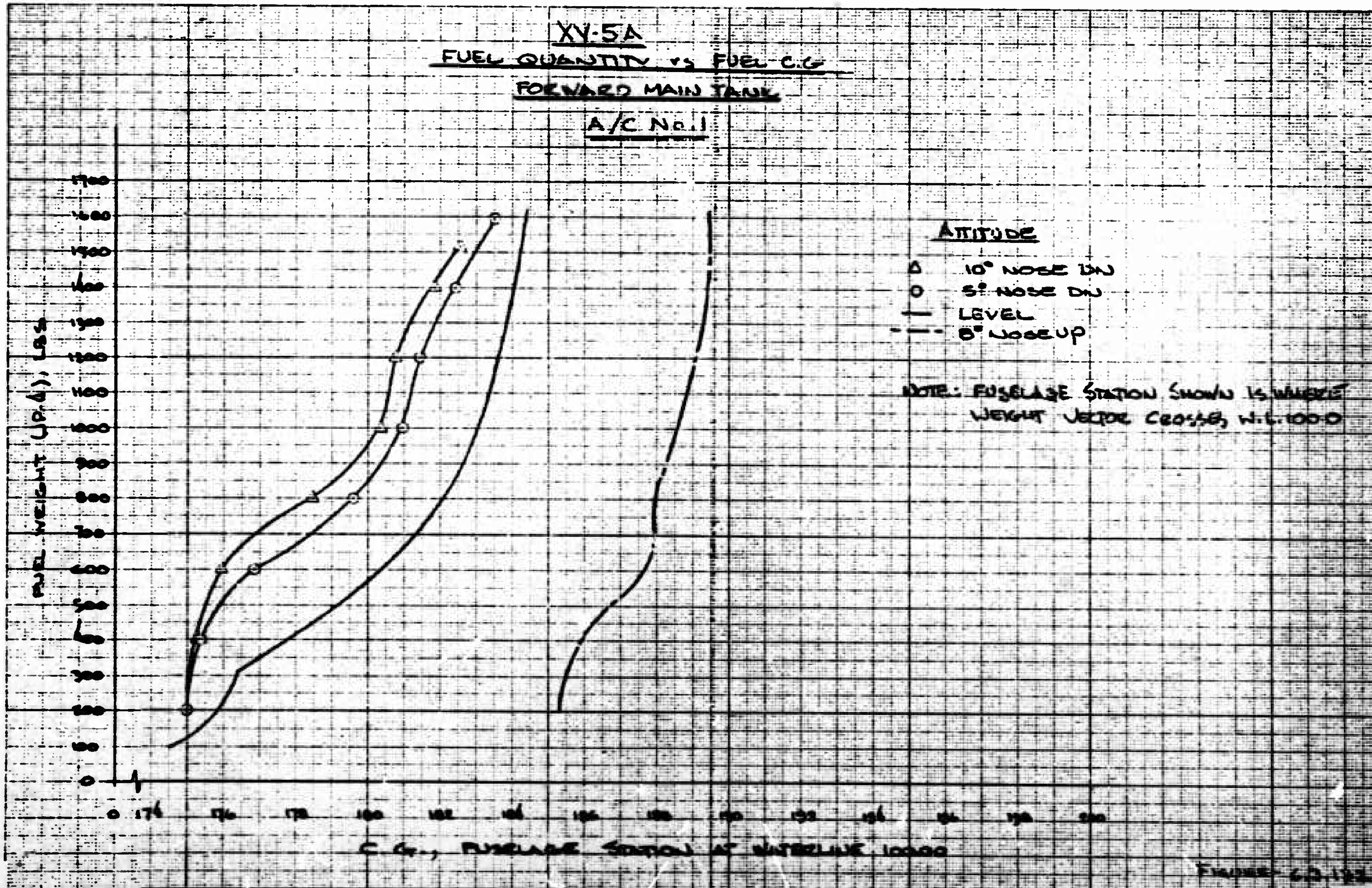


FIGURE- 6.3.131


XV-5A
FUEL QUANTITY vs FUEL C.G.
FORWARD MAIN TANK
A/C No. 1
ATTITUDE
10° NOSE DN
5° NOSE DN
LEVEL
8° NOSE UP
NOTE: FUSELAGE STATION SHOWN IS WHERE WEIGHT VECTOR CROSSES W.L. 100.0
FUEL WEIGHT (JP-4), LBS.
C.G., FUSELAGE STATION AT WATERLINE 100.00

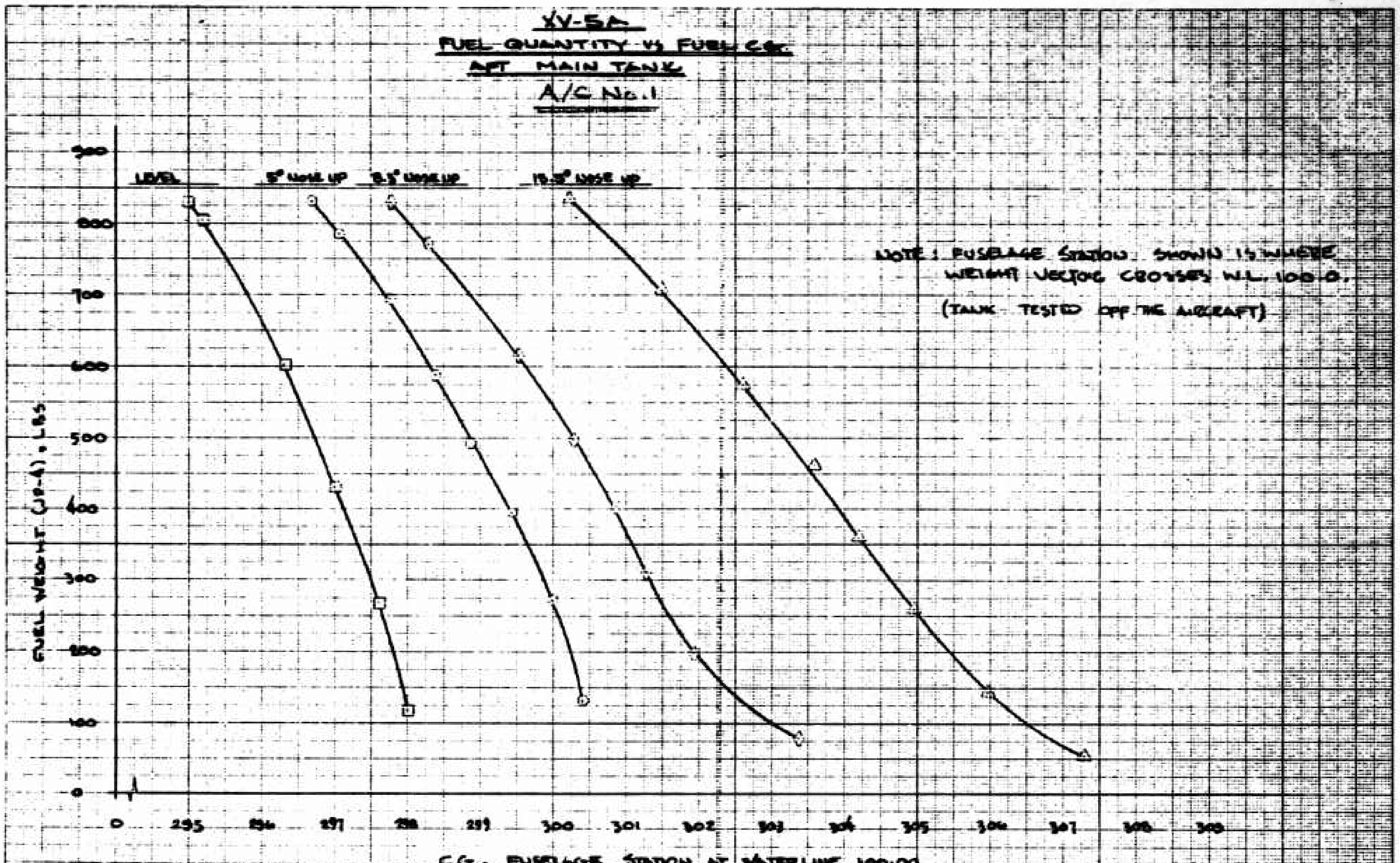


FIGURE 6.2.132

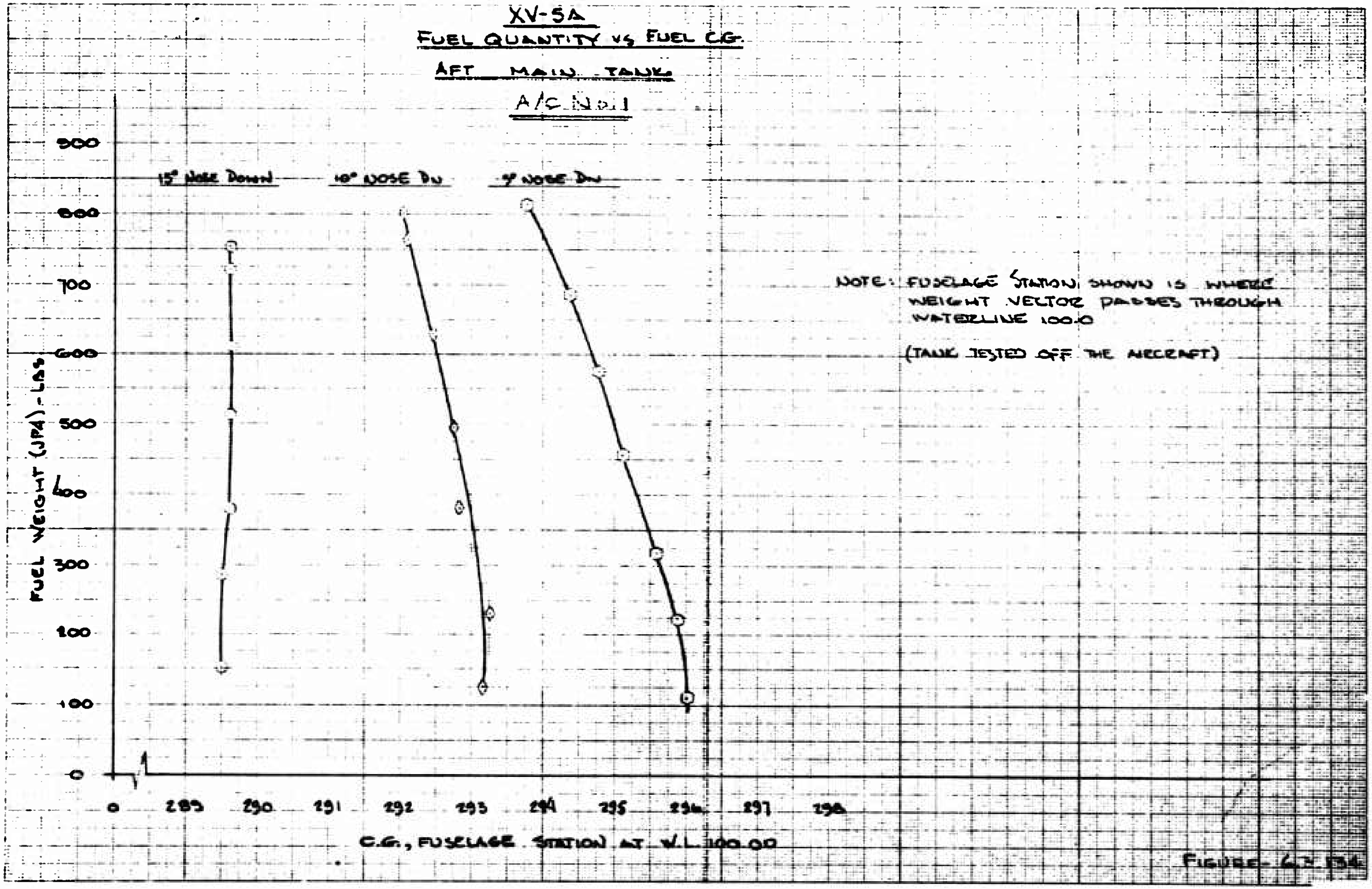


FIGURE 6.7.134

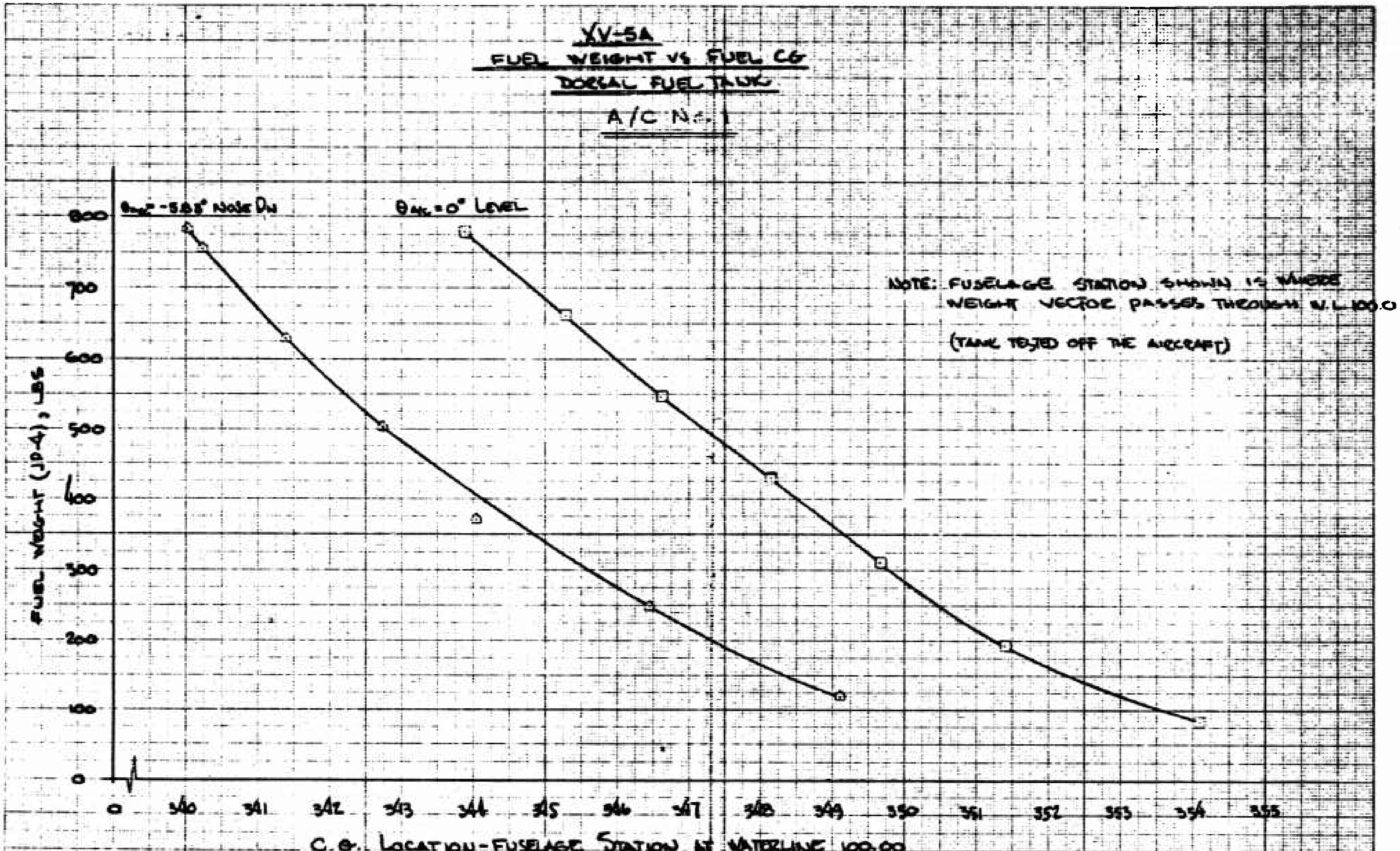
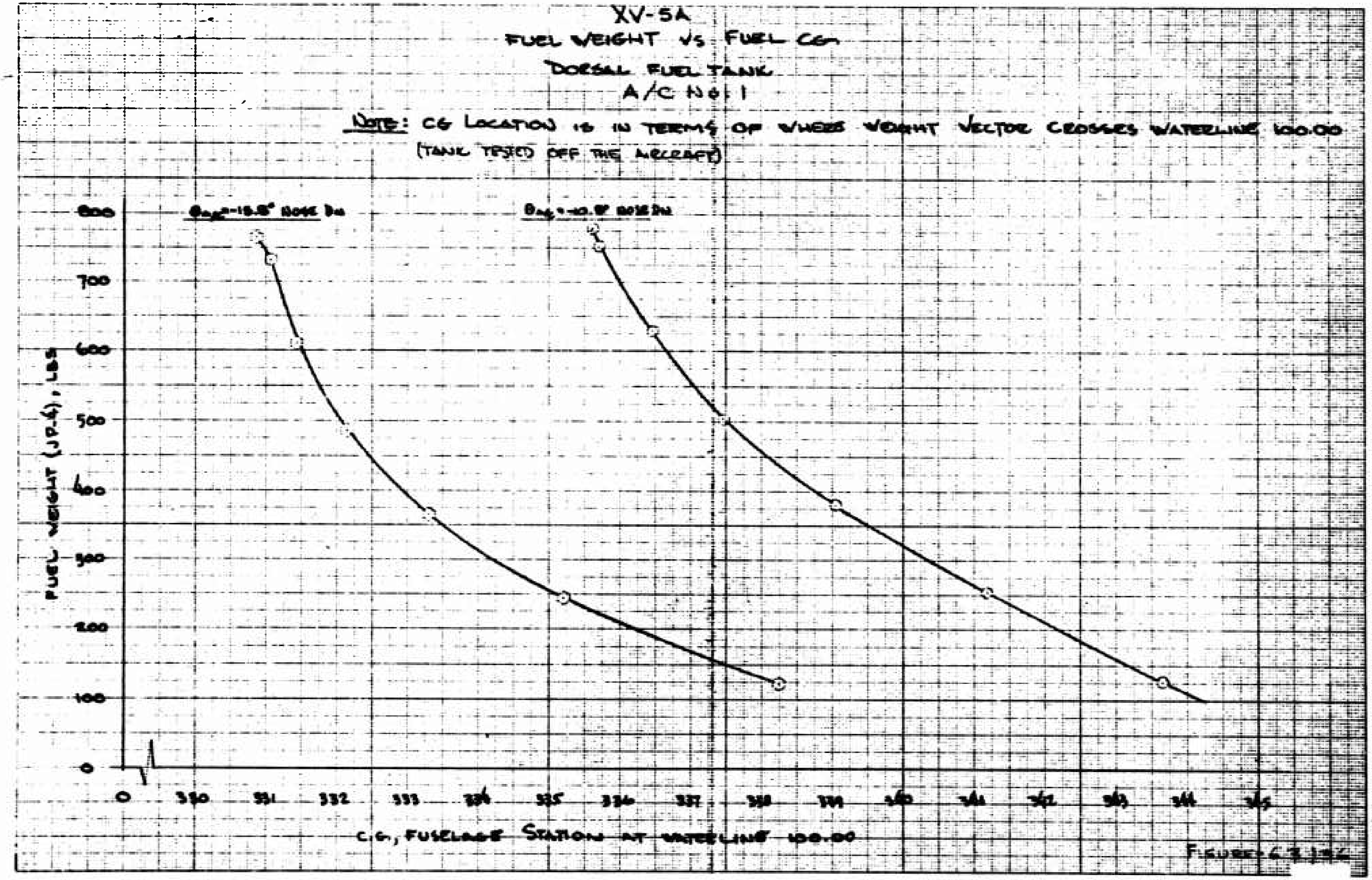
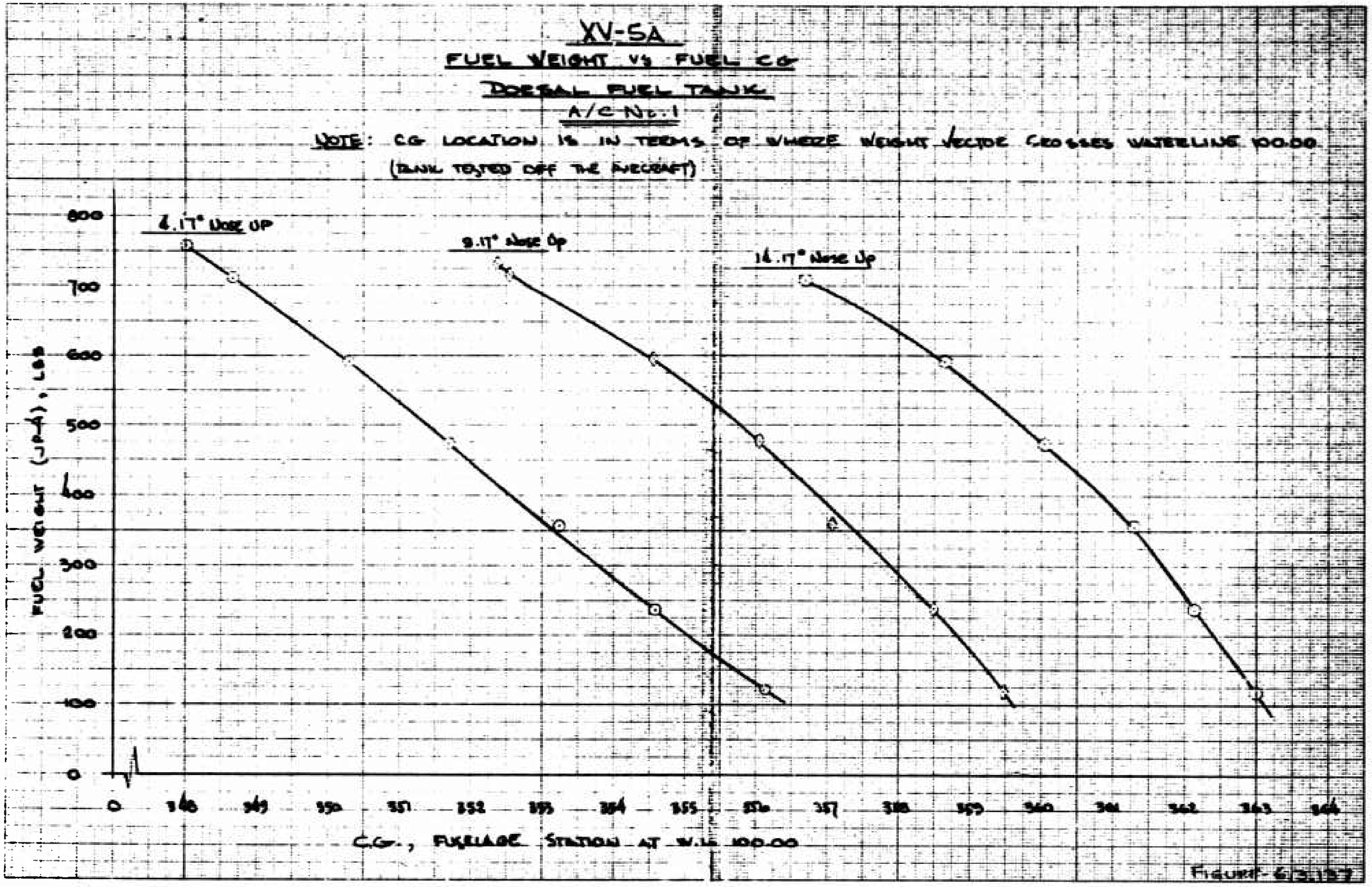


FIGURE 6.3.135


XV-5A
FUEL WEIGHT VS FUEL CG
DORSAL FUEL TANK
A/C NO. 1
NOTE: CG LOCATION IS IN TERMS OF WHERE WEIGHT VECTOR CROSSES WATERLINE 100.00
(TANK TESTED OFF THE AIRCRAFT)
FUEL WEIGHT (JP-4), LBS
800
700
600
500
400
300
200
100
0
330
331
332
333
334
335
336
337
338
339
340
341
342
343
344
345
C.G., FUSELAGE STATION AT WATERLINE 100.00



XV-5A
FUEL WEIGHT VS FUEL CG
DORSAL FUEL TANK
A/C No. 1
NOTE: CG LOCATION IS IN TERMS OF WHERE WEIGHT VECTOR CROSSES WATERLINE 100.00
(TANK TESTED OFF THE AIRCRAFT)
4.17° NOSE UP
9.17° NOSE UP
14.17° NOSE UP
FUEL WEIGHT (JP-4), LBS
0
100
200
300
400
500
600
700
800
0
348
349
350
351
352
353
354
355
356
357
358
359
360
361
362
363
364
C.G., FUSELAGE STATION AT W.L. 100.00

7.0 APPENDIX

7.1 RYAN REPORT NO. 63B102 AS ADDENDUM

XV-5A
INSTALLED SYSTEMS FUNCTIONAL
TEST PROCEDURE

RYAN

AERONAUTICAL COMPANY

REPORT NO. 63B102

24 OCTOBER 1963

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Product Test

Approved by: B. E. Browne

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ILLUSTRATIONS

RYAN 63B102

TABLES

DATA SHEETS

1.0 INTRODUCTION

This report details the installed systems functional testing to be accomplished on the U. S. Army Model XV-5A Lift Fan Flight Research Aircraft. Submittal is made in accordance with Government Contract No. DA44-177-TC-715 and General Electric Company purchase order 203-00727. Testing will proceed after all factory installations have been completed; after aircraft ground resonance tests, and before delivery of the No. 2 aircraft for flight tests.

The testing as presented here in detail is outlined in Ryan Report 62B075, "BASIC GROUND AND WIND TUNNEL TEST PROGRAM VZ-11" dated 1 June 1962. The complete test procedures as outlined in this report will be accomplished at the Ryan Aeronautical Company plant, Lindbergh Field, San Diego.

2.0 PURPOSE AND SCOPE

The purpose of these tests is to demonstrate that the XV-5A aircraft systems function in accordance with the design requirements.

The testing procedure is divided into 12 major tests. The order of appearance is the desired chronological order.

When the aircraft is received for functional tests, the hydraulic and pneumatic systems will have been flushed, filled and bled in accordance with Ryan Report 14359-6. The controls will have been rigged in accordance with Ryan Report 14359-5.

3.0 TEST PROCEDURES

3.1 ELECTRICAL SYSTEM CHECKOUT

The installed systems functional test procedure for the aircraft electrical system is outlined below. Required instrumentation and power may be seen in Figure 1. Accomplish the following steps:

Step 1. Pull all circuit breakers to OFF position.

Step 2. Move switches to positions as shown below:

Battery	OFF
Generator, LH	OFF
Generator, RH	OFF
Inverter #1	OFF
Inverter #2	OFF
Radio	OFF
Landing Gear	DOWN
Landing Gear STOL Select	STOL
Wing Flaps	NEUTRAL
Fire Extinguisher L Engine	OFF
Fire Extinguisher R Engine	OFF
Fire Extinguisher Cross Feed Select	NORMAL
Forward Fuel Tank Shut-Off Valve	OFF
Aft Fuel Tank Shut-Off Valve	OFF
Fuel Tank Cross Feed	CLOSED
Forward Boost Pump	OFF
Aft Boost Pump	OFF
Louver Lockout	LOCKOUT
Emergency Stabilizer Select	OFF
Thrust Spoiler	CLOSE
Stabilization Augmentation	OFF
Conversion System Changeover	PRIMARY
Mode Selector Switch	CTOL
Engine Ignition, RH	OFF

Engine Ignition, LH	OFF
Engine Anti-Ice, RH	ON
Engine Anti-Ice, LH	ON

NOTE: In the following test, observe indication of circuit overload by circuit breakers popping out.

Step 3. Connect external 28 VDC power supply to external power receptacle.

Step 4. Energize external power supply to supply 28 VDC power to airplane.

Step 5. Push in the following circuit breakers:

Generator Warning
DC Bus Control
Warning Test

Check the following conditions:

- Emergency bus, essential bus and non-essential bus monitor lights should light.

- Master caution light should light, and L and R generator OFF panels should light on annunciator panel.

- Press master caution light. It should go out.

- Press annunciator test switch. Master caution light should light, and all panels on annunciator panel should light.

- Release switch. Master caution light should go out and all panels, except L and R generators, on annunciator panel should go out.

Step 6. Push in the following circuit breakers:

#1 Inverter Warning
#2 Inverter Warning

Master caution light should light, and #1 and #2 inverter OFF panels should light on the annunciator panel.

Press master caution light; it should go out.

Step 7. Push in the following circuit breakers:

#1 Inverter Control
#2 Inverter Control
#1 Inverter Power
#2 Inverter Power

Move #1 and #2 inverter switch to the ON position. Number 1 and #2 inverter OFF panels should go out on annunciator panel. Check and record #1 and #2 inverter voltage, current and frequency.

Step 8. Push in the following circuit breakers:

#1 Hydraulic Pressure Indicator
#2 Hydraulic Pressure Indicator
L Oil Pressure Indicator
R Oil Pressure Indicator

The above hydraulic pressure and oil pressure indicators should move from the OFF position to the "O" pressure position.

Step 9. Push in the fuel quantity circuit breaker. The fuel quantity indicator pointers should start to move and should indicate the amount of fuel in the tanks within 50 seconds.

Step 10. Push in the left and right fuel flow indicator circuit breakers. The fuel flow indicators should move from the OFF position to the "O" fuel flow position.

Step 11. Push in the turn and bank indicator circuit breaker. It should be possible to hear the turn and bank indicator gyro coming up to speed.

Step 12. Push in the attitude indicator circuit breaker. The attitude indicator OFF flag should disappear.

Step 13. Push in the fire detector, left and right, and the structural overheat detector circuit breakers. Press the fire detector test switch. The left and right engine fire warning lights should light. Release of the switch should cause the lights should go out.

Press the structural overheat test switch. The master caution light should light and the structure overheat panel of the annunciator panel should light. Release of the switch should cause the lights to go out.

Step 14. Push in the radio circuit breaker. Turn radio switch to ON position. The radio should become energized. Turn radio OFF.

Step 15. Push in the fuel control cross feed, aft shut-off, forward shut-off, aft boost and forward boost circuit breakers.

Step 16. Push in the fuel warning, L press, R press, aft boost and forward boost circuit breakers in that order. When the L press circuit breaker is energized, the master caution light should light, and the L engine fuel pressure low panel on the annunciator panel should light. Press the master caution light; it should go out, but the annunciator panel should remain lighted. As each of the other circuit breakers indicated above is energized, the master caution light and the associated panel light should light. It is necessary to press the master caution light to extinguish, but the annunciator panel lights should remain lighted.

Step 17. Push in fuel warning level aft, and level forward circuit breakers. The associated fuel low-level warning lights should light (if fuel tanks are empty).

Step 18. Push in the low hydraulic pressure warning circuit breaker. The master caution light should light and the #1 and #2 hydraulic system low pressure warning panels should light on the annunciator panel.

Press the master caution light. It should go out and the annunciator panels remain lighted.

Step 19. Push in the landing gear low air pressure warning, position indicator, and control circuit breakers. The landing gear position indicators should roll to indicate that the L and R main gear and the nose gear are down and locked. The STOL warning light should light.

If there is no pressure in the landing gear emergency air pressure bottle, the master caution light should light and landing gear emergency low air pressure warning panel on the annunciator panel should light.

Press the master caution light. It should go out. The annunciator panel light should remain lighted.

Step 20. Push in the left and right engine ignition circuit breakers. Energize the left engine ignition switch momentarily. It should be possible to hear the ignitor plug in the left engine popping. Repeat the procedure for the right engine ignitor.

Step 21. Push in the conventional roll, pitch and yaw circuit breakers and the horizontal stabilizer trim indicator circuit breaker.

Move the CTOL yaw trim switch (on LH console) to left yaw and hold until position indicator indicates full left yaw position.

Move the CTOL yaw trim switch to the right yaw position and hold until position indicator indicates full right yaw position.

Energize left yaw until indicator indicates rudder trim tab is centered.

Energize trim switch on stick grip to right wing down position and hold until indicator shows full

RYAN 63B102

right wing down trim. Repeat for left wing down and then return tab to center position.

Step 22. Push in flaps drive and control circuit breakers, flaps/spoiler circuit breaker and the aileron droop circuit breaker. Move flap switch to the down position. Flaps should move to the full down position and the ailerons should move to the full droop position. The flap position indicator should follow the flaps to the full down position. Move flap switch to up, flaps and ailerons should return to up position.

Step 23. Push in left and right engine anti-ice circuit breakers. Move left engine anti-ice switch to OFF position; it should be possible to hear the solenoid click. Repeat the procedure for right engine anti-ice.

Step 24. Prior to applying hydraulic power to the airplane and prior to checkout of the conversion, horizontal stabilizer and automatic stabilization systems, the following steps should be taken:

- Push in the remaining circuit breakers except spares.

- Move mode selector switch to VTOL and back to CTOL. This sets up the magnetic latching relays in the electrical flight control mixer box.

3.2 SURFACE GAINS AND HYSTERESIS

During the following test, ground hydraulic power and 28 VDC electrical power are required except for rudder pedal vs. rudder position test. For the required instrumentation and power, see Figure 2.

3.2.1 Conventional Flight (CTOL Mode)

A. Continuous curves of longitudinal stick position vs. elevator position and longitudinal stick position vs. longitudinal stick force are required. The force will be hand applied slowly in the cockpit, throughout the control range in both forward and aft directions. These data will be recorded on an X-Y recorder. Check for longitudinal control movement with the stick in the extreme right, and left lateral positions, and with the horizontal stabilizer at -5°, 0° and 20°. Ground hydraulic power will be required to position the horizontal stabilizer only.

B. Continuous curves of lateral stick position vs. aileron position and lateral stick position vs. lateral stick force are required. The force will be hand applied slowly in the cockpit throughout the control range in both right and left lateral directions. These data will be recorded on an X-Y recorder. Check for lateral control movement with the stick in the extreme forward and aft longitudinal positions.

C. Continuous curves of rudder pedal position vs. rudder position and rudder pedal position vs. rudder pedal force are required. The force will be applied slowly in the cockpit throughout the control range in both right and left rudder directions. These data will be recorded on an X-Y recorder.

3.2.2 Hovering Flight VTOL Mode $\beta_V = 0°$

A. A continuous curve of collective control position vs. pitch fan control door position is required. The collective control will be displaced slowly throughout the control range in both up and down directions. These data will be recorded on an X-Y recorder.

B. Continuous curves of collective control position vs. wing fan louver actuator positions and collective control position vs. collective control force are required. The force will be hand applied slowly in the cockpit throughout the control range in both the up and down directions. These data will be recorded on an X-Y recorder.

C. Continuous curves of longitudinal stick position vs. pitch fan control door actuator position and longitudinal stick position vs. longitudinal stick force are required. The force will be applied slowly in the cockpit throughout the control range in both longitudinal forward and aft directions. These data will be recorded on an X-Y recorder. Conduct at neutral collective, full-up collective and full down collective.

D. Continuous curves of lateral stick position vs. wing fan louver actuator positions and lateral stick position vs. lateral stick force are required. The force will be hand applied slowly in the cockpit throughout the control range in both right and left lateral directions. These data will be recorded on an X-Y recorder. Conduct at neutral collective, full up collective and full down collective.

E. Continuous curves of rudder pedal position vs. wing fan louver actuators position and rudder pedal position vs. rudder pedal force are required. The force will be foot applied slowly in the cockpit throughout the control range in both right and left rudder directions. These data will be recorded on an X-Y recorder. Conduct at neutral collective, full up collective and full down collective.

3.3 FLIGHT CONTROLS STABILITY

3.3.1 Lateral Stick to Aileron, CTOL Mode

For the required instrumentation and power, see Figure 3. The control stick will be excited at the grip in the lateral direction at a double amplitude of 0.30 inches about stick neutral. The vibration frequency will be increased from 1 cps (cycle per second) to 35 cps, maintaining a constant double amplitude. Stick input acceleration and aileron servo acceleration will be recorded on a Minneapolis/Honeywell Visicorder, Model 906A-168 FGH or equivalent.

3.3.2 Lateral Stick to Wing Fan Louver, VTOL Mode $\beta_V = 0°$

Repeat operations of paragraph 3.3.1, except record log amplitude ratio, $\frac{\text{wing fan louver actuator displacement}}{\text{lateral stick displacement}}$ and phase shift vs. vibration frequency. The response curves will be made by employing Industrial Measurements Model 100 Servo Analyzer and its associated X-Y recorder.

3.3.3 Longitudinal Stick to Pitch Fan Door Servo VTOL Mode $\beta_V = 0°$

The control stick will be excited at the grip in the longitudinal direction at a double amplitude of 0.15 in. about stick neutral. Record log amplitude ratio $\frac{\text{pitch fan door servo displacement}}{\text{longitudinal stick displacement}}$ and phase shift vs. vibration frequency as per paragraph 3.3.2.

3.3.4 Rudder Pedal to Wing Fan Louver, VTOL Mode $\beta_V = 0°$

A rudder pedal will be excited at a double amplitude of 0.15 inches about rudder pedal neutral. Record log amplitude ratio $\frac{\text{wing fan louver actuator displacement}}{\text{rudder pedal displacement}}$ and phase shift vs. vibration frequency as per paragraph 3.3.2.

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3.3.5 <u>Collective Control to Wing Fan Louvers Frequency Response, VTOL Mode $\beta_V = 0°$</u>

For the required instrumentation see Figure 4.

The collective control will be excited at the grip at a double amplitude of 0.30 inches about mid-position. The vibration frequency will be increased from 1 cps to 35 cps, maintaining a constant double amplitude while recording log amplitude ratio $\frac{\text{wing fan louver actuator displacement}}{\text{collective control displacement}}$ and phase shift vs. vibration frequency. The response curves will be made employing Industrial Measurements Model 100 Servo Analyzer and its associated X-Y recorder.

3.3.6 <u>Step Inputs</u>

Mechanical stops will be installed on the stick, collective control and rudder pedals to limit their travels to approximately 50% of normal full travel. Recordings of step inputs and responses will be made on a Minneapolis Honeywell Visicorder, Model 906A-168 FGH or equivalent.

3.3.6.1 Conventional Flight (CTOL Mode)

For the required instrumentation and power, see Figure 5.

A. The stick will be moved by hand as rapidly as possible in the longitudinal forward and aft directions while simultaneously recording longitudinal stick position and elevator position vs. time.

B. The stick will be moved by hand as rapidly as possible in the lateral right and left directions while simultaneously recording lateral stick position and aileron position vs. time. The test will be done twice; once with both hydraulic systems, and once with #1 hydraulic system only.

C. The rudder will be moved as rapidly as possible by foot in the right and left rudder direction while simultaneously recording rudder pedal position and rudder position vs. time.

3.3.6.2 Hovering Flight (β_V = 0°) (VTOL Mode)

For the required instrumentation and power, see Figure 6.

A. The collective control will be moved by hand as rapidly as possible in the up and down direction while simultaneously recording collective control position, pitch fan door actuator position, and wing fan louver actuators position vs. time.

B. The stick will be moved as rapidly as possible in the longitudinal forward and aft direction while simultaneously recording longitudinal stick position and pitch fan door actuator position vs. time.

C. The stick will be moved as rapidly as possible in the lateral right and left directions while simultaneously recording lateral stick position and wing fan louver position vs. time.

D. The rudder pedals will be moved as rapidly as possible in the right and left rudder directions while simultaneously recording rudder pedal position and wing fan louver actuators position vs. time.

3.4 FLIGHT MODE CONVERSION SEQUENCE

For the required instrumentation and power, see Figure 7. The MLG shall be in the CTOL (wheels forward) position, the STOL Normal switch in the STOL position, and the electrical system changeover switch in "primary" position.

The following functions will be recorded on a Minneapolis Honeywell Visicorder, Model 906A-168 FGH or equivalent:

1. Vectoring actuator position.
2. Wing fan louver servo position.
3. Pitch fan door servo position.
4. Pitch fan inlet louver position.
5. Diverter valves end position signals.
6. Wing fan inlet door open, close and lock signals.
7. Flap and aileron positions.
8. Horizontal stabilizer position.

3.4.1 CTOL to VTOL Conversion Sequence

To be assured that the aircraft electrical systems are in CTOL mode, apply 28 VDC electrical power first and then hydraulic power to the aircraft, close all circuit breakers and check the following controls: Flaps-louver control in flaps up, louver-lockout switch in lockout, and mode selector switch in CTOL. Observe whether aircraft is in CTOL or VTOL mode. If in CTOL mode, proceed with step "A".

If in VTOL mode, set the following controls: Louver-lockout in coupled, and mode selector in VTOL. Energize the vector actuator to the 50° (stop) position. Set the mode selector in CTOL. The aircraft should convert to CTOL. Move flaps control to flaps up and louver-lockout to lockout.

Perform the following steps:

A. Recorder On

B. Flaps Louver Switch - Down (Flaps Down)

Allow approximately 15 seconds for flap operation. (Leave the flaps switch in the down position.)

C. Louver Lockout Switch - Coupled

Allow approximately 12 seconds for pitch fan inlet louver, wing fan door latch, and vectoring actuator operation. Check for cockpit "No-Go" indicator lamp to extinguish.

D. Mode Selector Switch - VTOL

Allow approximately 5 seconds for the conversion to complete.

E. Recorder Off

3.4.2 VTOL to CTOL Conversion Sequence

To be assured the aircraft electrical systems are in the VTOL mode, apply hydraulic power and 28 VDC electrical to the aircraft, close all circuit breakers and check the following controls: Flap-louver control in Down (flaps down), louver lockout switch in coupled, and mode selector switch in VTOL. The vector angle should be between -5° and +50°. Run the vector angle to -5° then up to 50° (stop) position.

Perform the following steps:

A. Recorder On

B. Mode Selector Switch to CTOL

Horizontal stabilizer moves down to -5°; diverter valve moves to CTOL, and wing fan inlet doors close.

C. <u>Flaps-Louver Control to Flaps Up</u>

The following should occur: Wing fan doors will latch, pitch fan louvers close, pitch fan control doors close, the wing fan louvers close, the thrust vector indicator moves to full close (87°), and flaps will move to up position (approximately 15 seconds for all functions to occur).

D. <u>Recorder Off</u>

Repeat paragraph 3.4.1 B. through D. Set louver-lockout switch to coupled. Set mode selector switch to CTOL and operations of Part B above should occur.

E. <u>Recorder On</u>

F. <u>Set Louver Lockout Switch to Lockout</u>

Operations of Part C above should occur, <u>except</u> for flaps operation.

G. <u>Recorder Off</u>

3.4.3 <u>Emergency Conversion</u>

A. Repeat the testing of paragraphs 3.4.1 and 3.4.2 with the #1 hydraulic system pressurized only, and then with the #2 hydraulic system pressurized only.

B. Repeat part A with the electrical system changeover switch in "Standby" position

3.4.4 <u>Conversion Abort Test</u>

Perform the conversion sequence in paragraph 3.4.1 down to Step D. Move the mode selector switch to VTOL, wait approximately 1 second then move the mode selector switch back to CTOL position. The aircraft should immediately reconvert back to CTOL mode.

Perform the conversion sequence in paragraph 3.4.2 down to Step B. Move the mode selector switch to CTOL, wait approximately 2 seconds then move the mode selector switch back to VTOL position. The aircraft should immediately re-convert back to VTOL mode.

3.5 COCKPIT CHECKOUT

For the required instrumentation power, see Figure 8.

3.5.1 Pitot System

3.5.1.1 Leakage and Flow Test

Disconnect both the 10 to 150 knot airspeed indicator and the 50 to 650 knot airspeed indicator. Plug the lines using AN 806 plugs. The aircraft shall be in CTOL mode.

Disconnect and remove the SCD T0001 pitot tubes, and connect a pressure pump to the impact pressure (143K008-43) line.

Apply 20 psig (pounds per square inch, gage) to the system and leak check all joints, fittings and unions in the pitot system. There shall be no leakage.

Reduce the applied pressure to 5 psig. Remove the AN 806 plugs in the cockpit to be certain of free flow to the instruments. Reduce the applied pressure to zero psig and replace the line plugs.

3.5.1.2 Automatic Functions

The aircraft should be in VTOL mode with the LND. gear STOL - normal switch in the STOL position. Apply 28 VDC electrical power and 3000 psi hydraulic power to the aircraft.

Apply 5 psig pressure to the pitot system. Remove the plug from the 10 to 150 knot airspeed indicator. There should be air flow out. Convert to CTOL mode and air should not flow out of the 10 to 150-knot airspeed indicator line. Lower the applied pressure to zero psig and replace the line plug.

Slowly increase the applied pressure. At or before 1.90 in. Hg (inches of mercury), the landing gear differential pressure switch should actuate. Check switch actuation using an ohmmeter. An open circuit between pins C_2 and NC_2 indicates an air speed above 150 knots.

Also at 1.90 in. Hg, pull the pilot's seat speed sensor arming key; the speed sensor plunger should move from left to right. Push plunger back and re-install key. Increase pressure to 2.67 in. Hg. The speed sensor should change phase. Pull the speed sensor arming key, and the plunger should not move from left to right. Re-install key.

Slowly decrease pressure. At 1.27 in. Hg, the seat speed sensor should change phase. Pull the key and the plunger should move from left to right. Re-install key.

At 1.056 Hg, the landing gear differential pressure switch should actuate (there should be continuity between Pins C_2 and NC_2).

3.5.2 Static System

3.5.2.1 Leakage and Flow Test

Disconnect the altimeter and the rate of climb indicator. Plug the lines using AN 806 plugs. The lines to the 10 to 150-knot and 50 to 650-knot airspeed indicator shall remain plugged.

Connect the pressure pump to the static pressure (143K008-45) line. Apply 20 psig to the system and leak check all joints, fittings and unions in the static system. There shall be no leakage.

Reduce the applied pressure to 5 psig and remove the AN 806 plugs in the cockpit to be certain of free air flow to the instruments. Reduce the applied pressure to zero psig and replace the line plugs.

3.5.2.2 Automatic Functions

Connect a vacuum pump to the static pressure (143K008-45) line. Slowly decrease the pressure. At 1.90 in. Hg. vacuum, the landing gear differential pressure switch should actuate. Check switch actuation using an ohmmeter. An open circuit between pins C_2 and NC_2 indicates proper functioning.

At 1.056 in. Hg, pull the speed sensor arming key; the key sensor plunger should move from left to right. Push plunger back and re-install key. Increase vacuum to 2.67 in. Hg. The speed sensor should change phase. Pull the speed sensor arming key. The plunger should not move from left to right. Re-install key.

Slowly decrease pressure. At 5.03 in. Hg vacuum, the landing gear altitude sensing switch should actuate. Check switch actuation using an ohmmeter. An open circuit between Pins C and NC indicates an aircraft altitude of 1,000 feet above terrain, or greater.

Slowly reduce the vacuum to 1.27 in. Hg. The seat speed sensor should change phase. Pull the key and the plunger should move from left to right.

Slowly reduce the vacuum to 1.056 in. Hg. The landing gear differential pressure switch should actuate. There should be continuity between pins C_2 and NC_2.

Remove all line plugs, disconnect the pump, and hook up the instruments. Re-install the electrical plugs in the landing gear differential pressure switch and the landing gear altitude sensing switch. Re-install the SCDT0001 pitot tubes.

3.5.3 Power Quadrant and Engine Controls

Pull both throttles full back. Check to see that the collective control twist grip is full clockwise and that the engine's throttle controls are against the stops.

Move the left throttle to idle. The collective control twist grip should move approximately 60° counterclockwise.

Move the right throttle to idle.

Rotate collective control twist grip full counterclockwise. The left and right throttle handles should have moved to full throttle and the pointers on the fuel control boxes should indicate wide open. Check that the force required to operate the collective control twist grip does not exceed 7 in. lbs.

3.5.4 Canopy Latch

With the canopy handle in the CANOPY OPEN position, the hooks should be full up, and ready to receive the canopy.

Lower the canopy. The hooks should latch over the pins before the hooks start to move downward. The hook springs should compress approximately 1-1/4 inches and hold the canopy.

Move the canopy handle to the VENT position. A definite catch should be apparent and the handle should remain securely in that position. A 3-3/4 inches opening should exist between the top of the windshield and the canopy.

Move the canopy handle past the locked position until a definite latch is felt. Release the handle. It should return to the locked position and remain there with the canopy down tight on the seals. All four shear pins should have entered a minimum of 1/4-inch into the receivers.

3.5.5 Drag Anti-Spin Chute

Pull the chute handle straight out to stop. The plastic drag chute cap should be released from the tail. The pilot chute should be out and the main chute should be firmly attached after this operation.

Rotate the chute handle 90° counterclockwise and pull out to stop. The chute should have been jettisoned.

Replace the chute and re-rig with the chute handle full in.

3.5.6 Automatic Throttle Cutback

Apply 28 VDC electrical power and 3000 psi hydraulic power to the aircraft. Remove plug 109 from the Waugh AC/106 RPM Indicating and Limiting Control Box. Ground pin L in plug 109.

Move throttles to the 90% RPM point and convert the aircraft to VTOL mode. When the wing fan louvers reach 45°, the pointers on the engine fuel control boxes should automatically

cut back to approximately a 70% power setting. Using a spring scale, determine the force required to override the throttle cutback. Push the throttle cutback reset switch on the collective control handle. The pointers on the engine fuel control boxes should move back to 90%.

3.6 ENGINE RUN - TEMPERATURE SURVEY

For the required instrumentation and equipment, see Figure 9. For Power Plant Operating Limits, see Appendix.

All temperature recording instruments shall be recording five minutes before and after each test condition or until the temperature stabilizes. Thermocouple instrumentation shall be supplemented by the use of Pyrodyne temperature indicating stickers or their equivalent. The approximate locations of the sticker type temperature indicators and the predicted temperature ranges are shown in Table IV. A 28 VDC ground power electrical power supply will be required for these tests.

Malfunction shall be defined as temperatures in excess of those listed in Table III, or hardware failure. Impending or actual malfunction shall be cause for immediate shutdown of test. The test shall not be resumed without the permission of the cognizant engineer. Recorded or indicated temperatures exceeding the values in Table III shall constitute a malfunction. Approach cautiously each new set of operating conditions, such as power setting, lift fan mode, louver actuation, thrust spoiler operation, etc. Sound power phone communications shall be maintained between the cockpit and the ground monitor station at all times.

Following each step, the recorded data, the aircraft, the Pyrodyne temperature indicators, the hydraulic reservoir temperature and accumulator pressure will be inspected before resuming test.

The aircraft must be secured to the ground in both vertical and longitudinal fore and aft directions.

3.6.1 L.H. Engine Cranking Test, Turbojet Mode

3.6.1.1 Preliminary Cockpit Setup

Before attempting to motor or start the engines, check the following: MLG in CTOL position, Mode Selector Switch in CTOL mode, MLG STOL Override Switch in STOL, #1 & #2 inverters OFF, #1 & #2 generators OFF, Battery OFF, #1 & #2 engine fuel feed valves OFF, Cross Feed Valve OFF, #1 & #2 booster pumps OFF, ignitors OFF and Throttles

in OFF position. All circuit breakers will be OFF, except the following: Fuel warning - L press, R press, boost aft, boost fwd, level aft, level fwd. Fuel control-cross feed, shut off - aft and fwd, boost fwd, boost aft, warn - hydro press, and fire extinguishers.

Apply external 28 VDC power. Turn on #1 inverter and #2 inverter (listen for inverter operation); turn #1 (LH) and #2 (RH) booster pump switches ON and OFF (the solenoid operation noise should be audible); turn #1 (LH) and #2 (RH) engine fuel feed valves and cross feed valve switches ON and OFF (the motor operation noise should be audible). Check the fuel quantity gauge. It should indicate approximately 150 pounds fore and aft.

3.6.1.2 Ignition and Fuel Control Test

Remove the ignitors from both engines. Switch to ignition on in the cockpit and check for positive sparking in the ignitors. If positive sparking is evident, turn off the ignition switch in the cockpit and replace the ignitors in the engines.

Disconnect the fuel line to the L. H. engine overspeed governor and the fuel line from the top of the R. H. engine flow meter. Extend these lines to five-gallon containers. Turn on #1 and #2 engine fuel feed valves and the #1 & #2 boost pumps. Attach the ground startcart to the L. H. engine inlet. Start the ground startcart. When the L. H. engine is rotating at firing speed (2000 RPM), check to see that no fuel is flowing into the five-gallon containers. Advance the #1 throttle momentarily to the idle position. Fuel should flow into the #1 five-gallon fuel container. Move #1 throttle back to the off position, the fuel flow should stop completely. Repeat using the #2 throttle. Shut off the ground startcart and reassemble and torque the fuel lines.

3.6.2 L. H. Engine Ground Idle, Turbojet Mode

Turn on L. H. engine fuel boost pump and start the ground startcart. When the engine reaches 2000 RPM, advance the throttle to the idle position for no longer than 15 seconds. A solid mist of fuel should be observed discharging from the #1 tailpipe. Check the #1 fuel flow indicator, it should read

between 200 and 300 lb. /hr. fuel flow. Move the throttle to the off position and shut off the L. H. engine boost pump and the ground start cart. Allow at least 3 minutes for fuel drainage before proceeding with test.

Turn on L. H. engine boost pump, L. H. fuel feed valve and, fuel anti-icing. Start the ground start cart. When the engine speed reaches 2000 RPM, turn on the #1 (L. H.) ignition and advance the throttle to the idle (13°) position.

NOTE: If engine light-off does not occur before a fuel flow indication of 350 lb. /hr. shut down and allow at least 3 minutes between starting attempts to permit complete fuel drainage.

Repeat the starting procedure. If engine light-off does not occur as above shut down and examine the engine per G. E. maintenance manual for the J85-GE-5B engine.

Observe exhaust gas temp. (EGT) indication. A rapid rise in EGT during starts is normal. The range of EGT for successful starts is between 825°F to 1500°F (441°C to 816°C).

NOTE: The maximum EGT limit is 1650°F for two seconds. If the EGT exceeds 1500°F, move throttle to off position, turn off ignition, and turn off LH fuel boost pump. If fire is suspected, continue motoring the engine with the ground start cart and blow CO_2 into engine air intake.

When the engine speed reaches 6600 RPM (40%), shut off the ground start cart and the ignition switch. Allow the engine to stabilize at idle, which is 7920 RPM (48%) for one minute. During this minute, observe and note the readings on the following cockpit instruments: LH fuel flow, LH oil pressure, LH RPM, LH EGT, #1 hydraulic system pressure, and #1 generator voltage. (Record on Data Sheet 8.)

At the end of the one minute operation, move the throttle to OFF position; turn off LH fuel feed valve and fuel boost pump. Inspect the recorded data, the Pyrodyne temperature indicators and the aircraft before resuming the test.

3.6.3 R. H. Engine Cranking Test

Check the cockpit set up as per paragraph 3.6.1.1.

Apply plant air through a cut-off valve and regulator to the engine bleed lines upstream from the cut-off valves.

Attach the ground start-cart to the R. H. engine inlet. Start the ground start-cart. When the R. H. engine is rotating at firing speed (2000 RPM) increase the pressure to the engine bleed lines to 25 psi. Advance the R. H. throttle to the idle position for no longer than 15 seconds. A solid mist of fuel should be observed discharging from the #2 tail pipe. Check the #2 fuel flow indicator, it should read between 200 and 300 lb/hr fuel flow. Move the throttle to the off position, lower the engine bleed pressure to zero and shutoff the ground start-cart. Disconnect the plant air line to the engine bleed. Allow at least 3 minutes for fuel drainage before proceeding with test.

3.6.4 R. H. Engine Ground Idle, Turbojet Mode

Start the LH engine and bring up to ground idle per paragraph 3.6.2.

Repeat the starting procedure of paragraph 3.6.2 for the RH engine.

Shut down the LH engine per paragraph 3.6.2.

Allow the RH engine to stabilize at idle, 7920 RPM (48%), for one minute. During this minute observe and note the readings on the following cockpit instruments: RH fuel flow, RH oil pressure, RH RPM, RH EGT, #2 hydraulic system pressure, and #2 generator voltage. (Record on Data Sheet 8.) Stop the RH engine per paragraph 3.6.2 and inspect the recorded data; the pyrodyne temperature indicators and the aircraft before resuming the test.

3.6.5 Fuel System Functional Checks and Thrust Spoiler Operation, Turbojet Mode

Fuel the aircraft forward and aft tanks to approximately 850 lbs each.

Start both engines and bring up to ground idle RPM per paragraphs 3.6.2 and 3.6.4. Hold at ground idle for two minutes. During this two minutes do the following: Turn on fuel cross feed valve and turn off LH engine fuel boost pump and feed valve (LH boost pump low pressure warning lamp should light). Turn on LH engine fuel boost pump (LH boost pump low pressure warning lamp should extinguish). Turn on LH engine fuel feed valve and turn off RH engine fuel boost pump and feed valve (RH boost pump low pressure warning lamp should light). Turn on RH engine fuel boost pump (RH boost pump low pressure warning lamp should extinguish). Turn on RH engine fuel feed valve and turn off fuel cross feed valve. During these tests there should be no variation in operation in either engine.

Clear the sides of the aircraft of all obstructions and personnel. Extend thrust spoilers and then retract them. Shut down the engines and inspect the recorded data; the Pyrodyne temperature indicators and the aircraft before resuming the test.

3.6.6 Ground Idle, Diversion Test

Start the engines and bring up to idling speed. Check that the STOL override switch is in the STOL position. Select flaps down, louver-lockout in coupled, VTOL on mode selector switch and record as per paragraph 3.4. Maintain the engines at idle speed in lift fan mode for one minute maximum. Select CTOL on mode selector switch and record as per paragraph 3.4. Shut down the engines and inspect the recorded data; the Pyrodyne temperature indicators and the aircraft before resuming the test.

3.6.7 Ground Idle, Lift Fan Mode, Longitudinal Stick Control

Start the engines and bring up to idling speed. Select VTOL on mode selector switch. Check that pitch, yaw and roll trim are at zero and the vector actuator is at zero. Move the stick slowly to full fore and aft position and back to neutral. Select CTOL on mode selector switch. Shut down the engines and inspect the recorded data; the Pyrodyne temperature indicators and the aircraft before resuming test. During this phase of the test, record longitudinal stick position,

pitch fan control door servo position and horizontal stabilizer position.

3.6.8 Ground Idle, Lift Fan Mode, Lateral Stick Control

Start the engines and bring up to idling speed. Select VTOL on mode selector switch. Check that pitch, yaw, and roll trim are at zero and the vector actuator is at zero. Move the stick slowly to full left and right position, and back to zero. Select CTOL on mode selector switch. Shut down the engines and inspect the recorded data, the Pyrodyne temperature indicators and the aircraft before resuming test. During this phase of the test, record lateral stick position and wing fan louver servo position.

3.6.9 Ground Idle, Lift Fan Mode, Collective Control

Start engines and bring up to idling speed. Select VTOL on mode selector switch. Check that pitch, yaw and roll trim are at zero and the vector actuator is at zero. Move the collective control to full up, full down and back to full up position slowly. Repeat with thrust vector at 30°. Convert back to CTOL. Shut down the engines and inspect the recorded data; the Pyrodyne temperature indicators and the aircraft before resuming test. During this phase of the test, record collective control position, pitch fan control door servo position and wing fan louver servo position.

3.6.10 High Power Engine Operation

Repeat the tests of Paragraph 3.6.2, 3.6.4, 3.6.5, 3.6.6, 3.6.7, 3.6.8, and 3.6.9 for the following engine speeds: 65% (10729 RPM) 75% (12390 RPM) 85% (10420 RPM) 90% (14850 RPM) 98% (16180 RPM) 99% (16180 RPM) and 100% (16500 RPM).

3.6.11 During the 90% (14580 RPM) diversion test the automatic throttle cutback system will be tested. Plug the 143G023 Fan Overspeed Test Inst. into the aircraft at ground test jack J215. Start the engines, bring up to speed and convert as per paragraph 3.6.6. After the wing fan louvers have reached 45° vector, actuate the override sensitivity switch

on the 143G023 test instrument. The engines speeds should automatically cut back to approximately 70% power (12000 RPM). Recheck at increasing vector angles from 10° up until throttle cutback occurs. (Design band 20° to 45°.)

NOTE: For Power Plant Operating Limits, see APPENDIX.

3.7 ENGINE RUN ELECTRICAL SYSTEMS CHECKOUT

For the required instrumentation and equipment see Figure 10.

The aircraft should be in CTOL mode (see paragraph 3.4) and the external hydraulic power disconnected.

Start both engines and set them to at least 75% RPM, and accomplish the following steps:

Step 1. Move L generator switch to RESET and then to ON; LH generator OFF panel on annunciator panel should go out. LH generator voltmeter should indicate 30 volts.

Step 2. Disconnect external 28 VDC power supply from external electrical power receptacle. LH generator ammeter should read approximately 90 amperes. Non-essential bus indicator light should go out.

Step 3. Move R generator switch to RESET and then to ON. RH generator OFF panel on annunciator panel should go out. RH generator voltmeter should indicate 30 volts. RH generator ammeter should show half of the average load that the two generators are carrying within ±5 amperes. LH generator ammeter should now read the same as RH, within 5 amperes of the average of the two. The non-essential bus indicator light should light.

Step 4. Move battery switch to ON position.

Move inverter No. 1 switch to OFF position. Master caution light should light and inverter No. 1 OFF panel on annunciator panel should light. Press master caution light; it should go out, but the panel should remain lighted. All items on No. 2, 115 VAC bus should cease to function.

Step 5. Move inverter No. 1 switch back to ON position. Inverter No. 1 OFF panel on annunciator panel should go out. Items on No. 2, 115 VAC bus should function again.

Step 6. Move inverter No. 2 switch to OFF position. Master caution light should light and inverter No. 2 OFF panel on annunciator panel should light. Items on No. 2, 115 VAC bus should cease to function. Press master caution light; it should go out.

Step 7. Move inverter No. 2 switch back to ON position. Inverter No. 2 OFF panel light on annunciator panel should go out. Items on No. 2, 115 VAC bus should again function.

Step 8. Move L generator switch to OFF position. Master caution light should light, and LH generator OFF panel on annunciator panel should light. Non-essential bus indicator light should go out. L generator ammeter should drop to zero. R generator ammeter should rise indicating that the R generator has picked up the load carried by the L generator. Press master caution light; it should go out leaving the annunciator panel light on.

Step 9. Move R generator switch to OFF position. Essential bus indicator light should go out. All items on the essential bus should cease to function. Master caution light should light and the RH generator OFF. No. 2 inverter off panels on the annunciator panel should light. Press master caution light; it should go out.

The battery should operate all items on the emergency bus for 10 minutes. All items on the emergency bus should be tested for proper operation including conversion during this period.

Step 10. Shut down engines and move battery switch to OFF position. All indicator lights and annunciator panel lights should go out.

NOTE: Replace battery with new or recharged unit.

3.8 AUTO-STABILITY TESTS

3.8.1 Test Preparation

For required instrumentation and equipment, see Figures 11 and 12.

All aircraft controls and switches shall be in the positions required for VTOL mode of flight operation. The wing fan louvers shall be trimmed to a zero Vector, zero Stagger position, and the pitch doors shall be trimmed to a neutral lift position with the stick and rudder pedals at neutral. All Auto-stabilizer wiring shall be connected per Ryan Drawing 143X002.

Determine that the Auto-stabilizer Primary and Standby AC and DC circuit breakers and Master Power switch are off.

Turn off Primary System switches on instrument panel.

Verify that Auto-stabilizer Hardover, Primary On and Standby On indicators are not illuminated.

3.8.2 System Power Check

Turn on Primary System AC circuit breaker and Master Power switch. Determine Primary gyro package operation by motor noise. (The Primary gyro package is located under the pilot's seat). Turn off Master Power switch and Primary AC circuit breaker.

Turn on Standby System AC circuit breaker and Master Power switch. Determine Standby gyro package operation by motor noise. (The Standby gyro package is located under the pilot's seat). Turn off Master Power switch and Standby AC circuit breaker.

Turn on Primary and Standby DC circuit breakers and Master Power switch. Either "Primary On" or "Standby On" indicator should be lit. Actuate "Auto-stabilizer" switch on the control stick and observe switching of indicators. Actuate "Auto-stabilizer" switch to obtain a "Primary On" condition. Turn off the Primary DC circuit breaker and observe switching to "Standby On" condition. Turn on Primary DC circuit breaker and observe return to "Primary On" condition.

3.8.3 Preliminary Gain Adjustment

Before Null and Phase adjustments can be made, the variable gain controls should be set to some nominal value. For purposes of this procedure, the Primary System gain controls on the instrument panel should be positioned at approximately mid-position. The Standby System gain controls, located on the front of the amplifier will be set during bench checkout.

3.8.4 Amplifier-Gyro Check

For the required instrumentation and power, see Figure 11.

Connect the Auto-Stabilizer test box as indicated in Table I. Connect the gyro outputs at the rate table to the aircraft gyro wiring by means of the Test Cable, P/N GTS-001.

Connect the test cable to the aircraft Primary gyro wiring. Operate the "Auto-Stabilizer" switch for "Primary On" operation.

3.8.4.1 Primary Roll Axis

Remove the Primary gyro package from its installation in the cockpit and install it on the rate table, in the roll sensitive axis. Rotate the table in a clockwise direction at a rate of 1.5 degrees per second. Adjust the "Roll Maneuver" gain on the instrument panel for approximately 8 ma (milliamperes) output current.

3.8.4.2 Primary Yaw Axis

Install the Primary gyro package on the rate table in the yaw sensitive axis. Rotate the table in a clockwise direction at a rate of 1.5 degrees per second. Adjust the "yaw maneuver" gain on the instrument panel for approximately 8 ma output current.

3.8.4.3 Primary Pitch Axis

Install the Primary gyro package on the rate table in the pitch sensitive axis. Rotate the table in a clockwise direction at a rate of 1.5 degrees per second. Adjust the "pitch

maneuver" gain on the instrument panel for approximately 8 ma output current.

Re-install the Primary gyro package in the aircraft.

3.8.4.4 Standby System

Connect the test cable to the aircraft Standby gyro wiring. Operate the "Auto-stabilizer" switch for "Standby On" operation. Remove the Standby gyro package from its installation in the cockpit and repeat the procedures of paragraphs 3.8.4.1, 3.8.4.2, 3.8.4.3 for the Standby system. The Standby "Maneuver" gain adjustments are located on the front of the amplifier chassis. Remove the Standby gyro package from the rate table, and re-install in the aircraft.

3.8.5 System Gain Adjustment

For the required instrumentation and power, see Figure 12. During this test, ground hydraulic power is required.

Connect the Auto-stabilizer test box as indicated in Table II.

The gain adjustments for the Primary system are located on the LH side of the instrument panel. Operate the "Auto-stabilizer switch to obtain "Primary On".

Apply an "In-Phase" 400 cps signal of .024V RMS to the Roll channel gyro input. Adjust the "Roll Hold" control on the instrument panel to obtain an 8 ma output. (This should give a wing fan louver deflection of 3.4 degrees. The louvers on the RH forward and LH aft actuators should move forward. The louvers on the RH aft and LH forward actuators should move aft.) Allow a minimum of 40 seconds for integrator action.

Displace the control stick laterally a minimum of 1/2 the total travel and hold in this position. Apply a .600V RMS signal and adjust the "Roll Maneuver" control on the instrument panel to obtain an 8 ma output. Holding the stick displaced, operate the "Roll" power switch on the instrument panel to OFF. The output current should go to zero. Return

the switch to ON and the output should read 8 ma. Allow the stick to return to neutral.

Apply an "In Phase" 400 cps signal of .024V RMS to the Yaw channel gyro input. Adjust the "Yaw Hold" control on the instrument panel to obtain an 8 ma on output. (This should give a louver deflection of 3.4 degrees. The louvers on the RH wing fan should move aft and the louvers on the LH wing fan should move forward.) Allow a minimum of 40 seconds for integrator action. Displace the rudder pedals a minimum of 1/2 the total travel and hold in this position. Apply a .600V RMS signal and adjust the Yaw "Maneuver" control on the instrument panel to obtain an 8 ma output.

Holding the pedals displaced, operate the "Yaw power switch on the instrument panel to OFF. The output current should go to zero. Return the switch to ON and the output should read 8 ma. Allow the pedals to return to neutral.

Apply an "In Phase" 400 cps signal of .024V RMS to the Pitch channel gyro input. Adjust the "Pitch Hold" control on the instrument panel to obtain an 8 ma output. (This should give a pitch door rotation of 8.75 degrees. The RH pitch door should rotate in a clockwise direction when looking aft from the front of the aircraft. The LH door should rotate in a counterclockwise direction.) Allow a minimum of 40 seconds for integrator action.

Displace the control stick longitudinally a minimum of 1/2 the total travel and hold in this position. Apply a .600V RMS signal and adjust the "Pitch Maneuver" control on the instrument panel to obtain an 8 ma output.

Holding the stick displaced, operate the "Pitch" power switch on the instrument panel to OFF. The output current should go to zero. Return the switch to ON and the output should read 8 ma. Allow the stick to return to neutral.

The gain adjustments for the Standby system are located on the front of the amplifier. Operate the "Auto-stabilizer" switch to obtain "Standby On".

Repeat the gain adjustment procedures for the Standby system Roll, Yaw and Pitch gains, respectively, adjusting the appropriate control on the front of the amplifier. Omit the portions regarding power switches, since they do not exist in the standby system.

3.8.6 System Functional Check

During this test, ground hydraulic power is required.

Remove the test box from the Auto-stabilizer wiring and connect all aircraft Auto-stabilizer wiring per Ryan Drawing 143X002.

Operate the "Auto-stabilizer switch to obtain "Primary On" operation.

With the ratio switch on the pilot's control box set to position 6, rotate the Primary gyro package to simulate an aircraft right roll condition. The wing fan louvers should move as described in paragraph 3.8.5.

Repeat for the yaw channel, simulating a right yaw condition. The wing fan louvers should move as described in paragraph 3.8.5.

Repeat for the pitch channel, simulating a pitch up condition. The pitch doors should move as described in paragraph 3.8.5.

Operate the "Auto-stabilizer" switch to obtain "Standby On" operation.

Repeat the check procedures for the Standby system.

Re-install gyro packages.

3.9 FAN FLIGHT TRIM RATES

For the required instrumentation and power, see Figure 13.

The Main Landing Gear shall be in the CTOL (wheels forward) position, the Landing Gear STOL-normal switch in the STOL position, and the mode selector switch in the VTOL position.

A. With the vector actuator at $\beta_V = 0°$, energize the pitch trim through a complete stroke. Record pitch fan door servo position vs. time. Energize the vector actuator to 15°, then increase the vector by 1° increments and energize pitch trim until the trim control transfers from pitch control door servo to the horizontal stabilizer. Record the pitch control door angle at the point of transfer.

B. With the vector actuator at $\beta_V = 30°$, operate the pitch trim through full horizontal stabilizer stroke. Record the horizontal stabilizer position vs. time. Determine the horizontal stabilizer rate for longitudinal stick displacements of ±3/4 inch.

C. With the vector actuator at $\beta_V = 0°$, energize the roll trim actuator through a complete stroke while recording wing fan louver servo positions vs. time. Energize the vector actuator to 15°, then increase the vector by 1° increments and energize the roll trim until the trim control transfers from wing fan louver servos to aileron tab. Record the vector angle at the time of transfer.

D. With the vector actuator at $\beta_V = 30°$, operate the roll trim aileron tab actuator through a complete stroke. Record aileron tab position vs. time.

E. With the vector actuator at $\beta_V = 0°$, energize the VTOL yaw trim actuator through a complete stroke. Record wing fan louver servos position vs. time.

F. With the vector actuator at $\beta_V = 0°$, energize the CTOL yaw trim actuator through a complete stroke. Record rudder tab position vs. time.

3.10 LANDING GEAR TESTS

For the required instrumentation and power, see Figure 14.

3.10.1 Brake Check (Aircraft on Jacks)

With a man stationed in the cockpit, spin the left-hand wheel which should be free to turn. Apply left wheel brake and check to see that wheel is locked by attempting to turn. Release left wheel brake and check to see that the wheel is free to turn. Repeat brake check for the right wheel. Check the clearance between the pads and disks which should be from .005 to .020 inch.

3.10.2 CTOL Main Landing Gear Functional Test (Aircraft on Jacks)

With the Main Landing Gear in CTOL position, up-latches unlatched, and ground safety pins in, check the following cockpit controls: Landing Gear handle - down, mode selector switch in CTOL. STOL override switch in normal. Apply hydraulic power and 28 VDC to aircraft ground power receptacles, then depress the following circuit breakers: Landing Gear Air, Landing Gear Warning, Landing Gear Indicator and Landing Gear Control. All gear position indicators should display "LOCKED DOWN". Remove ground safety pins and select "GEAR UP". The following sequence should occur: Gear indicators indicate "STRIPES", Main Landing Gear shall transfer to VTOL position, Main Landing Gear shall fully retract in 5 seconds maximum, up-latches shall latch, Main Landing Gear doors shall close and indicators shall display "UP". During this test, observe operation of nose gear.

NOTE: The red light in the gear handle will always illuminate when in the "UP" position unless the airspeed pitot head is pressurized to simulate 150 knots or greater airspeed.

After inspecting nose and Main Landing Gear doors for proper closure, select gear down. The following sequence shall occur: Indicators shall display "STRIPES", Main Landing Gear doors shall fully open, up-latches shall unlatch, Main Landing Gear will extend to VTOL position and then move to

CTOL position, and indicators shall display "LOCKED DOWN". During this test observe operation of nose gear.

3.10.3 VTOL Main Landing Gear Functional Test (Aircraft on Jacks)

Select VTOL mode on Mode Selector Switch. The Main Landing Gear indicators shall display "STRIPES", the Main Landing Gear will transfer to the VTOL position and indicators shall indicate "Locked Down". Retract and extend gear with Mode Selector Switch in VTOL position. The Main Landing Gear and nose gear should function as in CTOL position except that the Main Landing Gear will always stop and remain in the VTOL, wheels aft, position.

3.10.4 Downlock Override Check (Aircraft on Jacks)

Remove the micro-switch actuator arms from the lower torque links. Attempt to select UP on the Landing Gear control handle; the handle shall not move and the gear shall remain extended. Depress the DN LOCK RELEASE button above the Landing Gear control lever and simultaneously select UP with the lever. The Landing Gear shall retract normally and the doors shall close. Re-extend the Landing Gear and replace the micro-switch actuator arms.

3.10.5 STOL Override (Aircraft on Jacks)

Select STOL on STOL override switch and the following shall occur: STOL amber warning lamp shall illuminate, Main Landing Gear indicators shall display stripes, Main Landing Gear shall move to CTOL position, Main Landing Gear indicators shall display LOCKED DN. Then select "NORMAL" on STOL override switch: Main Landing Gear indicators shall display stripes, Main Landing Gear shall transfer to VTOL position, STOL warning lamp shall extinguish and Main Landing Gear indicators shall display LOCKED DN.

3.10.6 Emergency Pneumatic Extension System (Aircraft on Jacks)

3.10.6.1 No Load Functional Test

Install ground safety pins, remove outer door panels 143L008-1 and -2 and door idler links 143L031-13 and -15,

charge emergency system reservoir with dry nitrogen to 3000 psi. Check in cockpit for low pressure warning light OFF, and emergency control handle down with button flush with T-handle. Remove ground safety pins, select gear up. When gear is up and locked, support the Main Landing Gear and doors on a safety stand. Remove electrical plugs to Landing Gear and Landing Gear door hydraulic selector valves and remove safety stands. Select DN with Landing Gear control handle; there should be no door or gear action. Pull up on emergency control handle (handle should lock in the up position), Main Landing Gear doors shall fully open; then the Main Landing Gear and Nose Landing Gear shall unlatch, extend and lock down. Note any movement of Main Landing Gear before doors are fully open. Record emergency system nitrogen pressure from gage in Main Landing Gear bay. Insert Landing Gear ground safety pins and reconnect electrical plugs to Landing Gear and door selector valves.

Remove Landing Gear ground safety pins and return emergency control handle to down position. The emergency system pressure will vent audibly through system to the hydraulic reservoir. When the venting has stopped and the compressed nitrogen pressure in the door and gear cylinders has dropped below 40 psi, select gear up on Landing gear control handle. The gear should retract and doors close.

3.10.6.2 <u>Low Pressure Emergency Extension with Stimulated 180 Knots Air Drag</u>

A cable loading system will be rigged to provide 170 pounds constant rearward load on each leg of the Main Landing Gear during the last 30° of gear extending motion (see Figure 14.) In the retract position, support the Main Landing Gear and doors on a <u>safety stand</u> and remove electrical plugs from Landing Gear and door selector valves. Reduce the emergency system nitrogen pressure until LG EMERGENCY AIR PRESS LOW INDICATOR on annunciator panel illuminates (1600 psi). Recharge with nitrogen until light just extinguishes. Record pressure gage reading. Remove the Main Landing Gear safety stand. Select DN with Landing Gear control handle; there should be no door or gear action. Pull up on emergency control handle (handle should lock in the up

position), Main Landing Gear doors shall fully open, then the Main Landing Gear and Nose Landing Gear shall unlatch, extend and lock down. Note any movement of Main Landing Gear before doors are fully open. Record emergency system nitrogen pressure from gage in Main Landing Gear bay. Insert Landing Gear ground safety pins and reconnect electrical plugs to Landing Gear and door selector valves. Remove Landing Gear ground safety pins and return emergency control handle to down position. The emergency system pressure will vent audibly through system to the hydraulic reservoir. When the venting has stopped and the compressed nitrogen pressure in the door and gear cylinders has dropped below 40 psi, select Gear Up on Landing Gear control handle. The gear should retract and doors close.

Following this test, the 170 pound loading system will be removed; the outer door handle and idler links will be replaced in their correct location and the landing gear will be extended in CTOL position. Cycle the landing gear in the normal mode five times. Lock and insert ground safety pins. Recharge the emergency system nitrogen pressure and bleed the aircraft No. 2 hydraulic system. Remove the aircraft jacks.

3.10.7 External Hydraulic Power Directly to 2-Position Mode Change Actuator

Use a tractor, with brakes locked and tow bar, or similar method to restrain nose wheel and use a level paved area in line with airplane's centerline. Fasten a sling from a crane around the aft fuselage attachment point at Sta. 384, or use a cable from a concrete anchor to the Nose Landing Gear. The sling must allow for 2 feet fore and aft movement and ±1 foot vertical movement. The aircraft shall be loaded to 10,500 pounds with the center of gravity at Sta. 246. The Main Landing Gear wheels shall be free to turn. With no electrical power to the aircraft, connect the ground hydraulic power pressure line to the extend port of the mode change actuator and connect the return line to the retract port. Bleed pressure line hose. Apply hydraulic pressure to extend 2-position actuator slowly to VTOL position. Test for 2-position actuator internal lock function, using a continuity test between wires 1 and 2 from the actuator.

Reverse pressure and return lines. Apply hydraulic pressure to retract 2-position actuator slowly back to CTOL position. When actuator is fully retracted, check for operation of over-center lock mechanism. Remove tractor and tow bar from nose wheel.

3.10.8 MLG Wheels Restrained

Station a man in the cockpit to apply Main Landing Gear brakes. Extend and retract 2-position mode change actuator as accomplished in paragraph 3.10.7. At the conclusion of the test, reconnect the aircraft hydraulic lines to the mode change actuator. Connect the ground hydraulic power lines to the aircraft's ports and bleed the aircraft's hydraulic systems.

3.10.9 Hydraulic and Electrical Power to Aircraft

Check the following cockpit control: Landing Gear handle in down, mode selector switch in CTOL. STOL override switch in STOL. Also check to see that the ground safety pins are installed.

3.10.10 Nose Wheel Restrained

Attach the tractor with brakes locked and tow bar to the nose wheel. Apply electrical power to the aircraft and depress the following circuit breakers: Landing Gear Air, Landing Gear Warning, Landing Gear Indicator and Landing Gear Control. All gear position indicators should display "LOCKED DN" and the STOL amber warning lamp shall illuminate. The Main Landing Gear wheels shall be free to turn. Select VTOL on Mode Selector Switch. The Main Landing Gear will not change mode; however, all other fan flight controls will convert.

Select Normal on the STOL Override Switch. Indicators shall display "STRIPES", the Main Landing Gear wheels will transfer to the VTOL (aft) position and indicators shall display "LOCKED DN".

Select STOL on the STOL Override Switch. Indicators shall display "STRIPES", the STOL amber warning lamp shall

illuminate, the Main Landing Gear wheels will transfer to the CTOL (forward) lock position and indicators shall display "LOCKED DOWN". Remove tractor and tow bar from nose wheel.

3.10.11 Main Landing Gear Wheels Restrained

Check that nose wheel is free to turn and sling is in place. Set Main Landing Gear brakes and follow the switching procedure of paragraph 3.10.7.

3.11 CONTROLS PROOF LOADS

3.11.1 CTOL Mode (See paragraph 3.4.1)

For the required instrumentation see Figure 15.

Attach STZ-004 rudder tension regulator restraint fitting and STZ-005 elevator restraint fitting to the aircraft. The aileron boost servos will be hydraulically locked by capping the pressure and return ports with the units filled with oil. Calibrate the stick and rudder force and position instrumentation.

Apply the following loads in the cockpit with both hydraulic systems depressurized:

a. Lateral stick, left and right 100 pounds.

b. Longitudinal stick forward and aft 200 pounds.

c. Rudder pedals, left and right 300 pounds.

Record lateral stick position vs. stick force, longitudinal stick position vs. stick force, and rudder pedal position vs. pedal force on an X-Y recorder. Remove the rudder and elevator restraint fittings and reconnect the aircraft hydraulic lines to the aileron servos.

3.11.2 VTOL Mode (See paragraph 3.4.2)

The vector angle shall be at 0°, and the louver servos and the pitch control door servo will be hydraulically locked by capping the pressure and return parts with the units filled with oil. Calibrate the collective control position and force instrumentation.

Apply the previous loads shown in paragraph 3.11.1, Items a, b, and c to the stick and apply 150 pounds up and down to the collective control.

Record lateral stick position vs. stick force; longitudinal stick position vs. stick force; rudder pedal position vs. pedal force, and collective control position vs. collective force. Reconnect the aircraft hydraulic lines to the louver servos and the pitch control door servo.

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3.12 WEIGHTS - BALANCE AND FUEL TESTS

For the approved instrumentation and power, see Figure 16.

During this test, the Main Landing Gear shall be in the VTOL (Aft) position. The nitrogen charge in the oleos shall be released until they are bottomed. (Check to see that the shoulder on the oleo rod does not ride on the cylinder gland nut.) Replace the high pressure pneumatic fittings (MS 28889) with modified AN815-5 fluid fittings. Fill each oleo to 50% of total stroke (total stroke on nose oleo is 8.00 inches, Main Landing Gear oleo is 9.2 inches) using a hand hydraulic pump filled with MIL-H-5606. Fine leveling adjustment will be accomplished by adjusting the oleo lengths hydraulically during the test.

The Main Landing Gear will be blocked from rolling during all phases of this test. In addition, to eliminate the possibility of longitudinal tipping, an adjustable length safety line attached to concrete imbedded anchors will be tied to the nose Landing Gear during Nose-up tests and to the aft jack pad adapter during nose-down tests.

After each fuel test, and while the aircraft is still tilted for test, the unusable fuel will be measured. To prepare the aircraft, remove the tube assembly from the left and right fuel filters that lead to the engine. Attach lines from the filter outlet to the ground fuel supply tank. Remove the elements from the fuel filters and place in a clean sealed plastic bag.

To empty the fuel tanks, energize the engine fuel feed valves, but not the cross feed valve. Switch on both left and right boost pumps in the cockpit (28 VDC ground power is required). Apply approximately 25 psi air pressure into the engine bleed lines upstream from the cutoff valves. Run the pumps until the usable fuel has been pumped from the tanks. Shut off the engine fuel feed valves and the right and left boost pumps in the cockpit. Level the aircraft and drain the unusable fuel from the water drain taps at the bottom of the fore and aft main fuel tanks. Record the unusable fuel.

3.12.1 <u>Aircraft Attitude Zero Degrees (Use Data Sheet #1)</u>

Lift the aircraft, using a crane or jacks and place scales under the Main Landing Gear and the Nose Landing Gear wheels. Lower the aircraft onto the scales, and block the Main Landing Gear wheels. Level by placing a bubble type level on the Main Landing Gear door sill longeron and adjusting the oleo lengths. Attach and zero a longitudinal plumb-bob type protractor in the Nose Landing Gear area.

Record the weight appearing on the Nose Landing Gear and the Main Landing Gear minus the Main Landing Gear wheel blocks. Record scale readings for each 200 pound increment of fuel added to the forward main tank. Since the total capacity of the forward main tank is approximately 1703 pounds, the last increment will be = 103 pounds. At each fuel increment during the aircraft zero attitude test, record the indicated fuel from the gages in the cockpit. When first filling, note the quantity of fuel in the tank when the fuel low level amber light extinguishes on the cockpit fuel control panel.

Record the scale reading for each 200 pound increment of fuel added to the aft main and aft dorsal tanks. Since the total capacity of the aft main and aft dorsal tanks together is approximately 1689 pounds, the last increment will be approximately 89 pounds. At each fuel increment during the aircraft zero attitude test, record the indicated fuel from the gages in the cockpit. When first filling, note the quantity of fuel in the tanks when the fuel low level amber light extinguishes on the cockpit fuel control panel.

Empty all tanks and measure the unusable fuel as outlined in Paragraph 3.12 in preparation for the next phase of the test.

3.12.2 <u>Aircraft Attitude, 5° Nose Up (Use Data Sheet #2)</u>

Lift the aircraft using a crane or jacks, and replace the scales under the Nose Landing Gear with the GTS-005, fork lift adapter. Lower the aircraft with the scales under the Main Landing Gear. Using a fork lift truck, raise the Nose Landing Gear and set the GTS-006-3 stand under the fork lift

adapter. Lower the Nose Landing Gear and adjust the Nose Landing Gear oleo until the protractor indicates 5° nose up.

Record the weight appearing on the Main Landing Gear at zero fuel and each 200 pound increments as outlined in Paragraph 3.12.1.

Empty all tanks and measure the unusable fuel as outlined in Paragraph 3.12 in preparation for the next phase of the test.

3.12.3 <u>Aircraft Attitude, 10° Nose Up (Use Data Sheet #3)</u>

Raise the Nose Landing Gear using the fork lift and set the GTS-006-3 and GTS-006-2 stands under the fork lift adapter. Lower the Nose Landing Gear onto the stands and adjust the Nose Landing Gear oleo until the protractor indicates 10° nose up.

Record the weight appearing on the Main Landing Gear at zero fuel and each 200 pound increment as outlined in Paragraph 3.12.1.

Empty all tanks and measure the unusable fuel as outlined in Paragraph 3.12 in preparation for the next phase of the test.

3.12.4 <u>Aircraft Attitude, 15° Nose Up (Use Data Sheet #4)</u>

Raise the Nose Landing Gear using the fork lift and set the GTS-006-3, GTS-006-2 and GTS-006-1 stands under the fork lift adapter. Lower the Nose Landing Gear onto stands and adjust the Nose Landing Gear oleo until the protractor indicates 15° nose up.

Record the weight appearing on the Main Landing Gear at zero fuel and each 200 pound increment as outlined in Paragraph 3.12.1

Empty all tanks and measure the unusable fuel as outlined in Paragraph 3.12 in preparation for the next phase of the test.

3.12.5 Aircraft Attitude, 5° Nose Down (Use Data Sheet #5)

Place the scales on the GTS-004 Main Landing Gear stands using a crane. Lift the aircraft, using hoist 143G020 and set the Main Landing Gear on the scales, and the Nose Landing Gear on the fork lift adapter and stands as shown in Figure 16. Be certain safety lines are attached to the aircraft as outlined in Paragraph 3.12 and that the Main Landing Gear is blocked. Level the aircraft laterally by adjusting the Main Landing Gear oleo lengths. Adjust the Nose Landing Gear oleo length until the protractor indicates 5° nose down.

Record the weight appearing on the Main Landing Gear at zero fuel and each 200 pound increment as outlined in Paragraph 3.12.1.

Empty the tanks and measure the unusable fuel as outlined in Paragraph 3.12 in preparation for the next phase of the test. To level the aircraft longitudinally, lift the Nose Landing Gear using a fork lift truck and place GTS-006-3, GTS-006-2 and GTS-006-1 stands under the fork lift adapter. Lower the Nose Landing Gear onto the stands and adjust the Nose Landing Gear oleo until the protractor indicates zero degrees.

3.12.6 Aircraft Attitude, 10° Nose Down (Use Data Sheet #6)

Raise the Nose Landing Gear using the fork lift and set the GTS-006-3 stand under the fork lift adapter. Lower the Nose Landing Gear onto the stands and adjust the Nose Landing Gear oleo until the protractor indicates 10° nose down.

Record the weight appearing on the Main Landing Gear at zero fuel and each 200 pound increment as outlined in Paragraph 3.12.1.

Empty all tanks and measure the unusable fuel as outlined in Paragraph 3.12 in preparation for the next phase of the test.

3.12.7 Aircraft Attitude, 15° Nose Down (Use Data Sheet #7)

Lower the Nose Landing Gear and fork lift adapter onto the pavement. Adjust the Nose Landing Gear oleo until the protractor indicates 15° nose down.

Record the weight appearing on the Nose Landing Gear at zero fuel and each 200 pound increment as outlined in Paragraph 3.12.1.

Empty all tanks and measure the unusable fuel as outlined in Paragraph 3.12. Using a crane and hoist 143G020 lift the aircraft off the stands and lower to the pavement. Replace the tube assemblies to the fuel filters and re-install the fuel filter elements. Drain the oleos of oil, re-install the H/P pneumatic fittings (MS28889) and re-inflate per vendor's instructions (Loud Service Note No. 1510 LA TP-1).

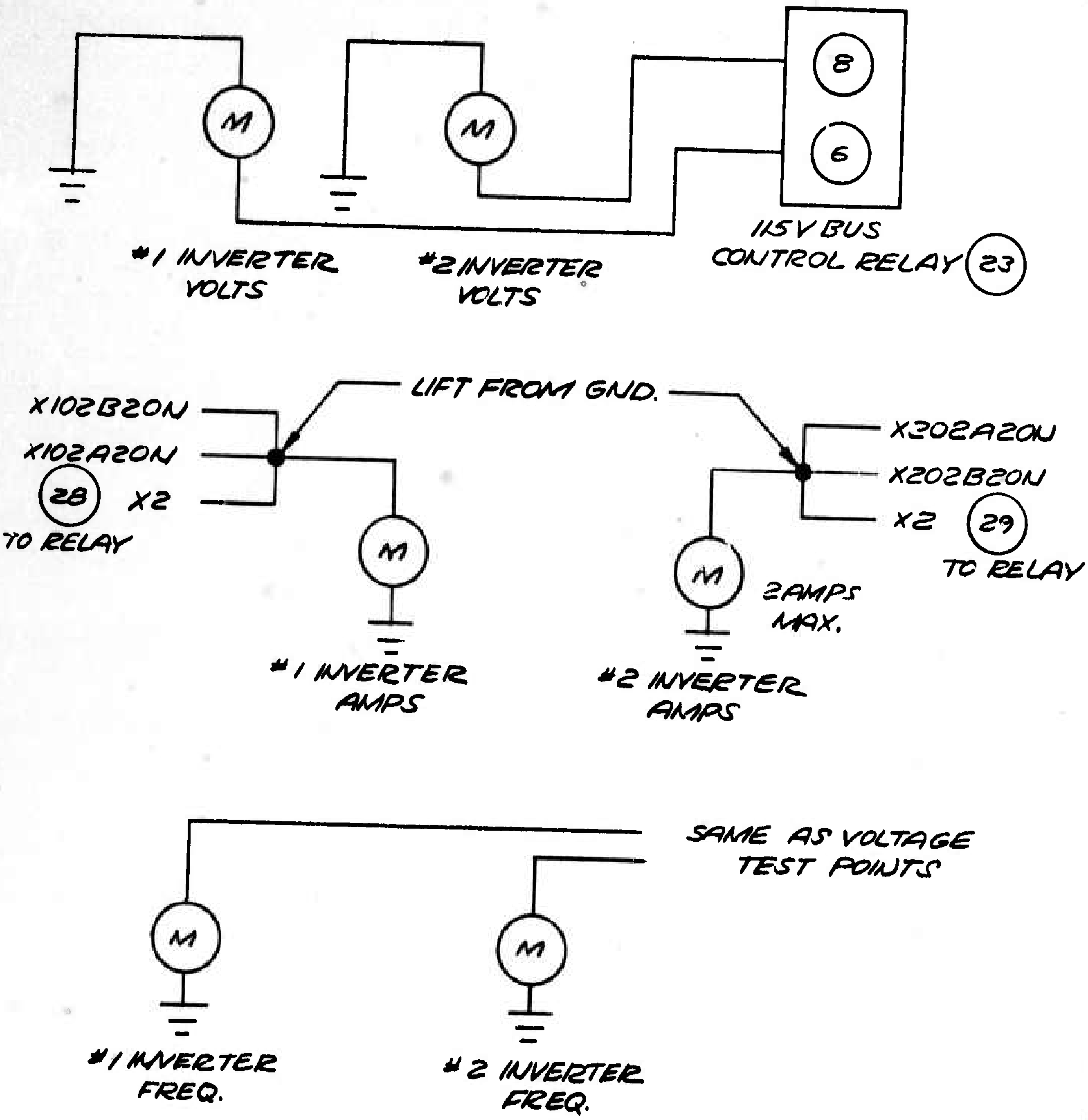

Figure 1 Electrical System Checkout Instrumentation

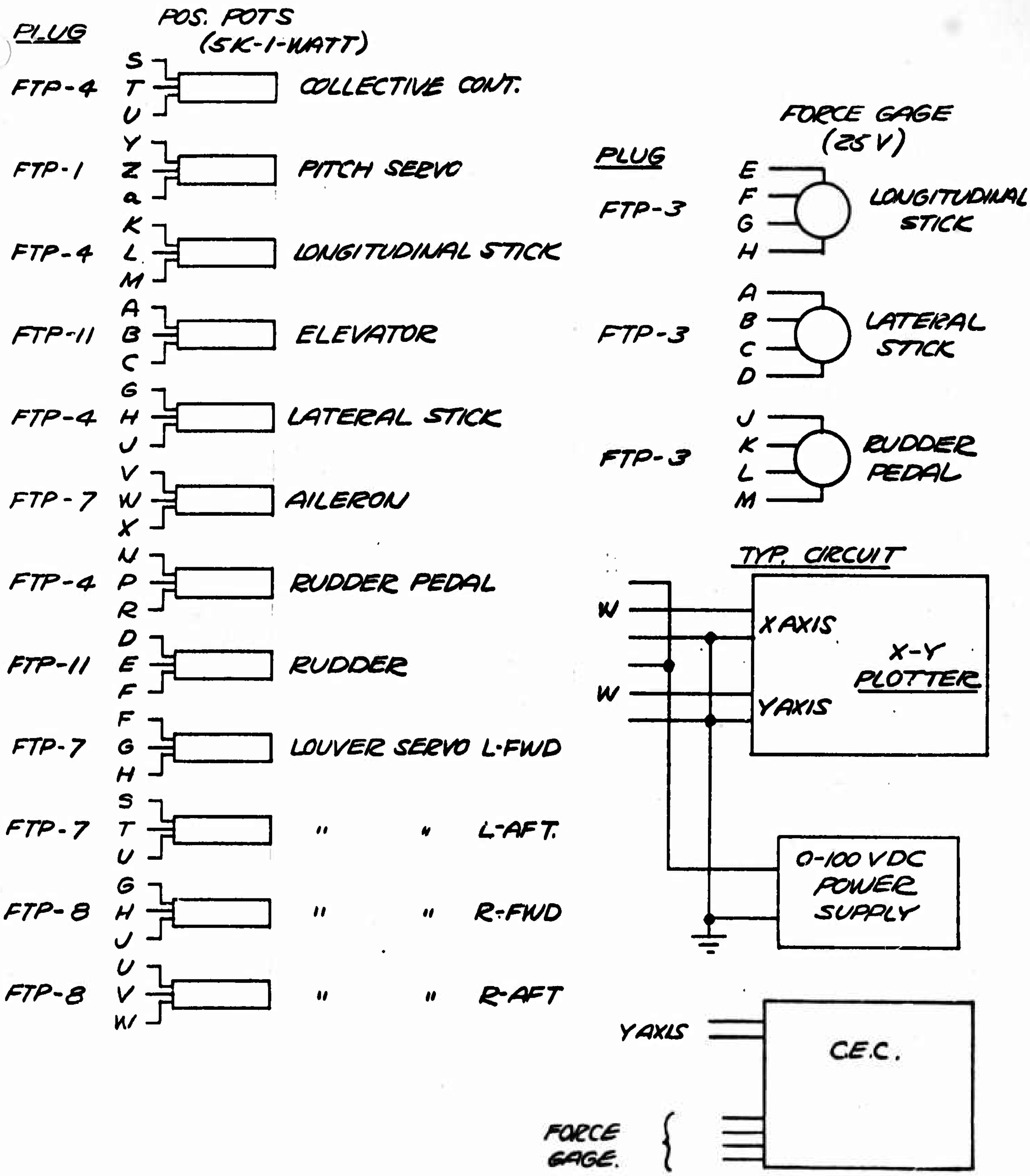

Figure 2 Surface Gains and Hysteresis Instrumentation

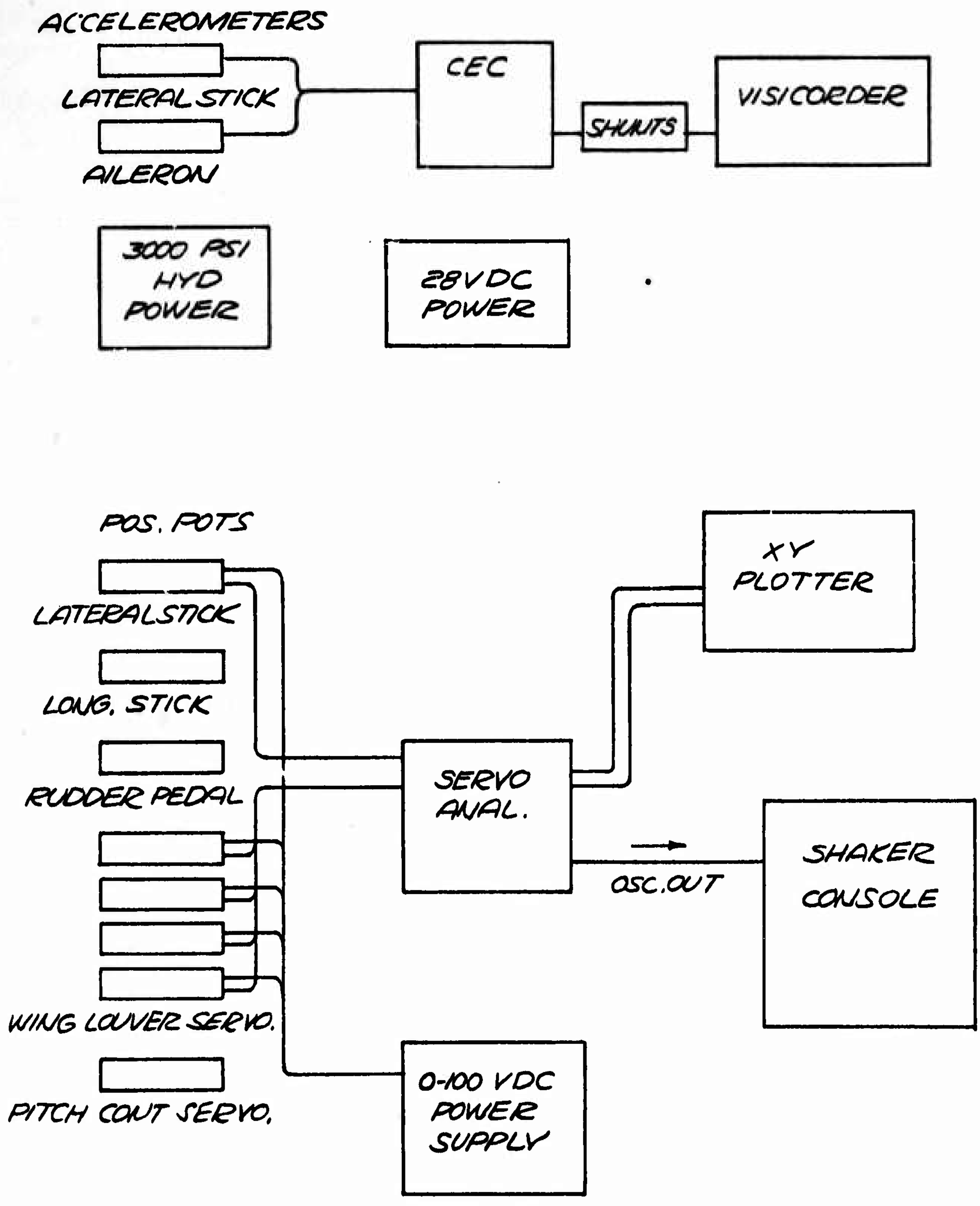

Figure 3 Flight Controls Stability Control Stick and Rudder Pedal Frequency Response

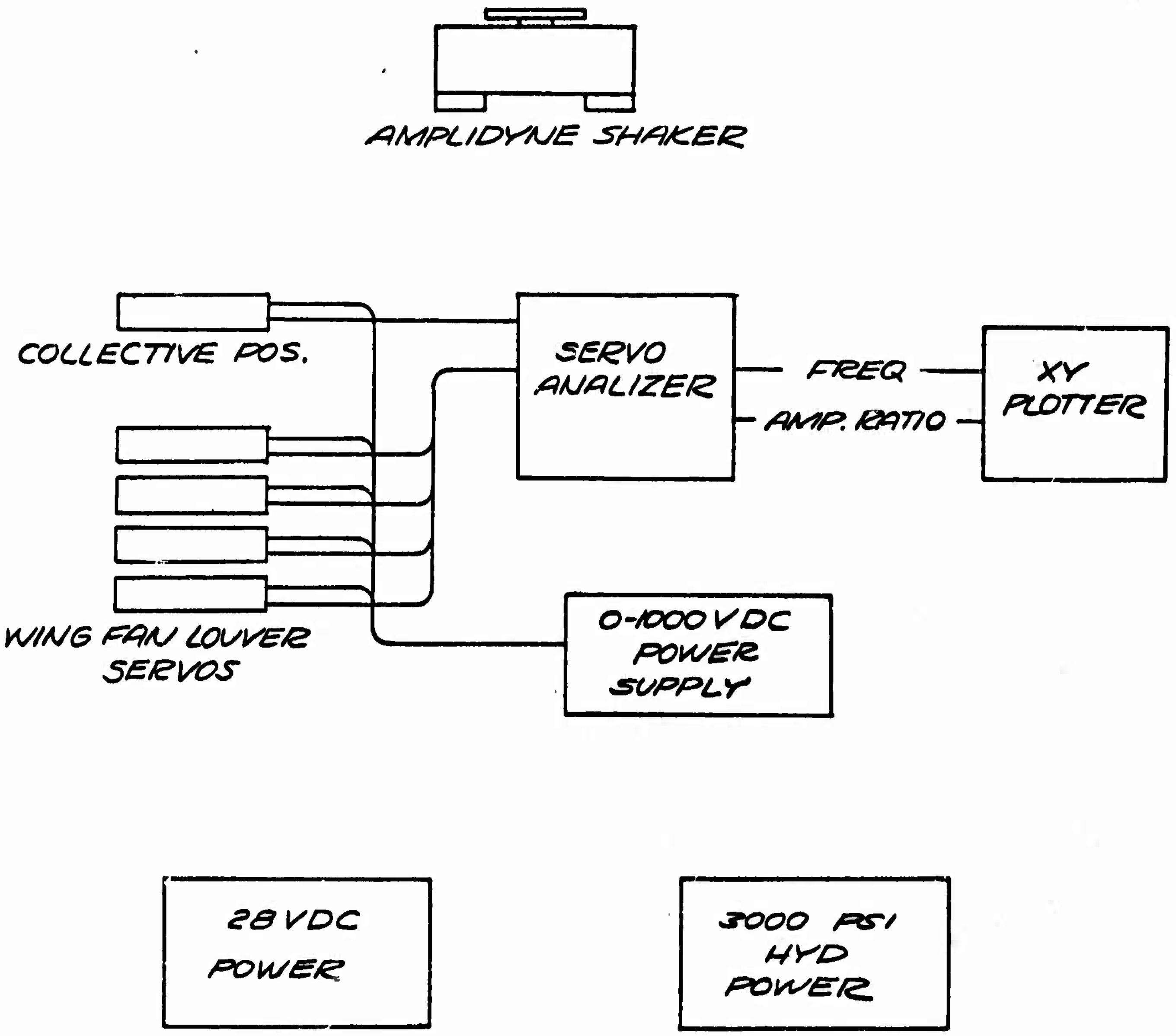

Figure 4 Flight Controls Stability Collective Frequency Response

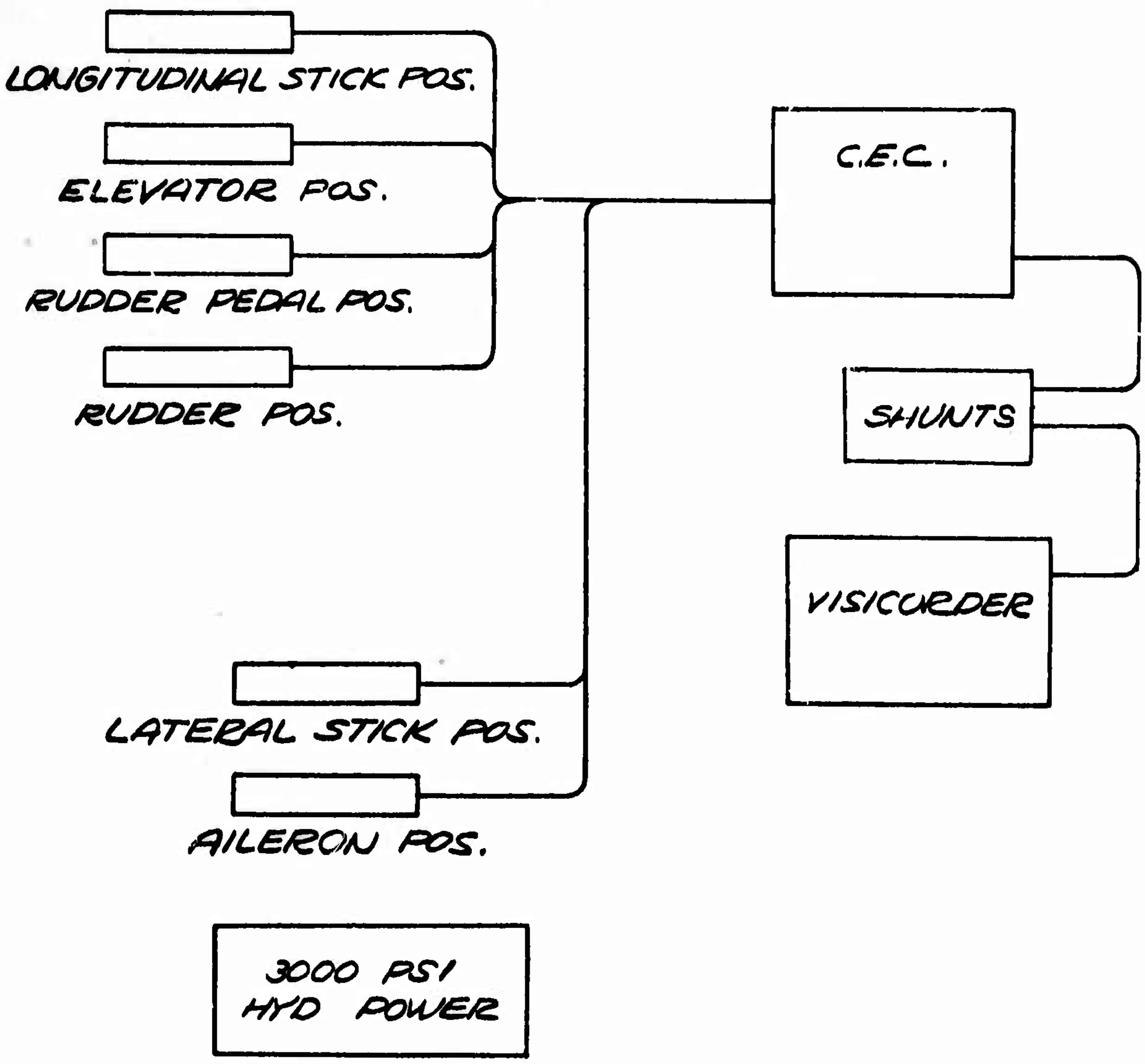

Figure 5 Flight Controls Stability CTOL Mode, Step Input

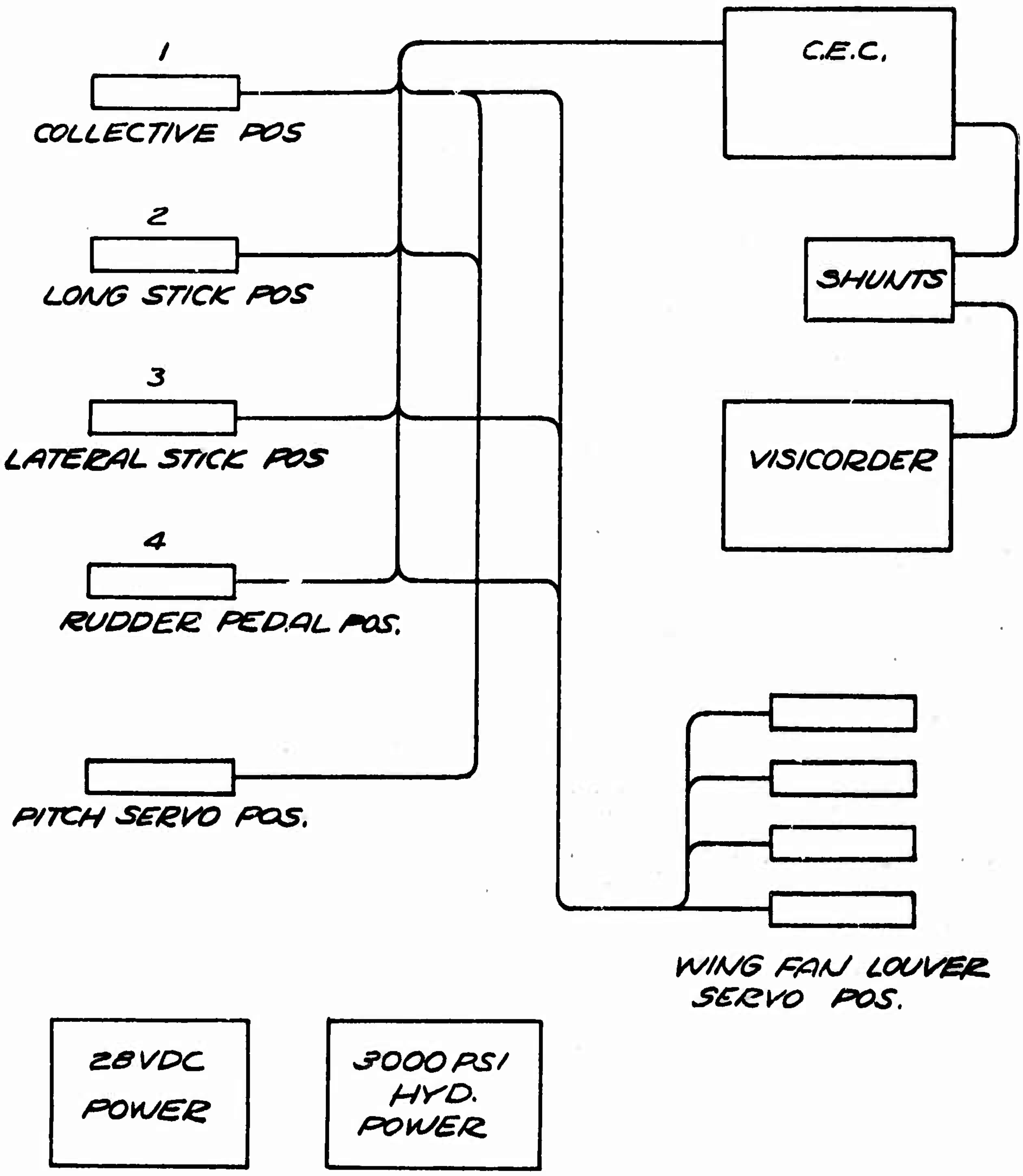

Figure 6 Flight Controls Stability VTOL Mode, Step Input

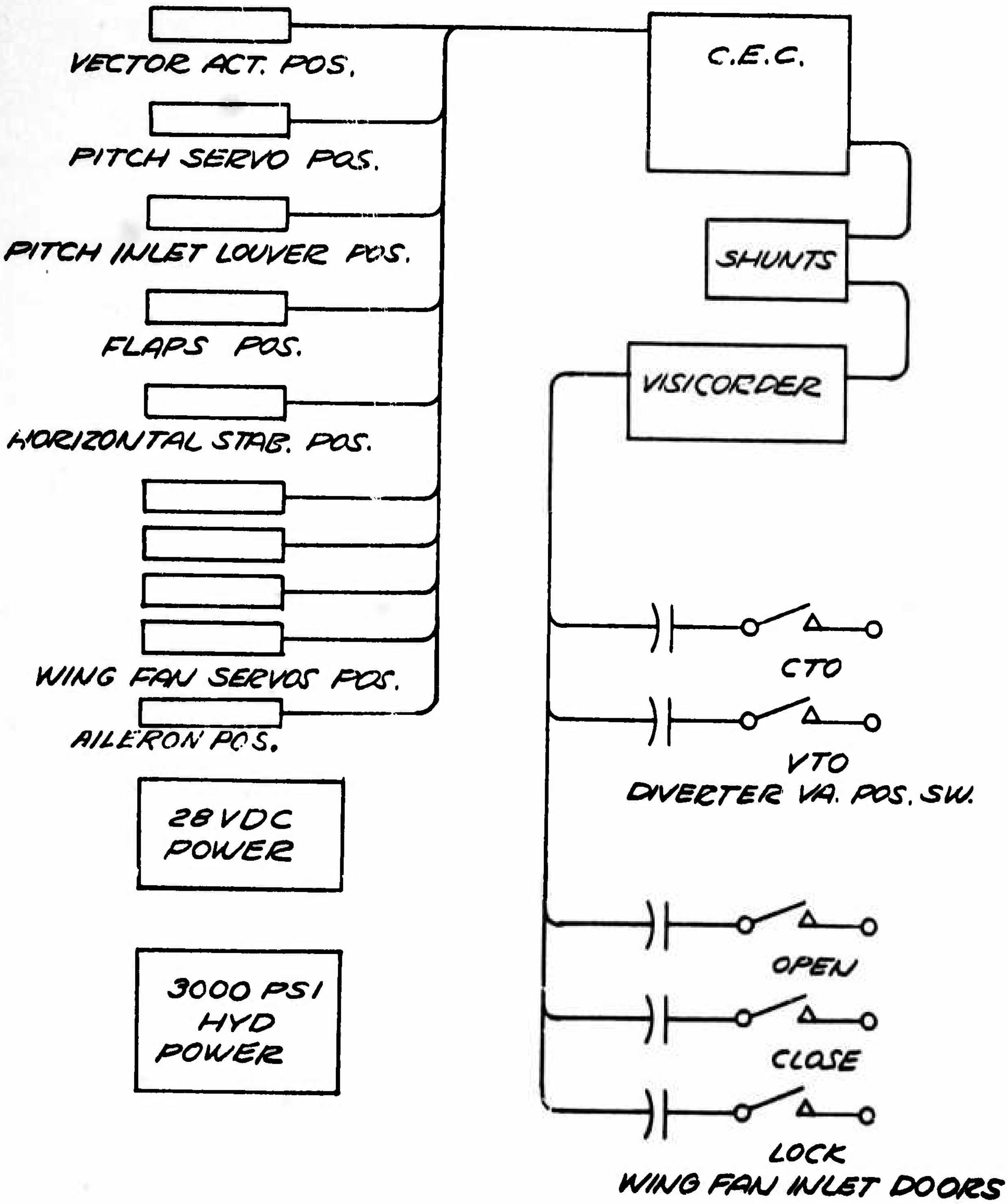

Figure 7 Flight Mode Conversion Sequence

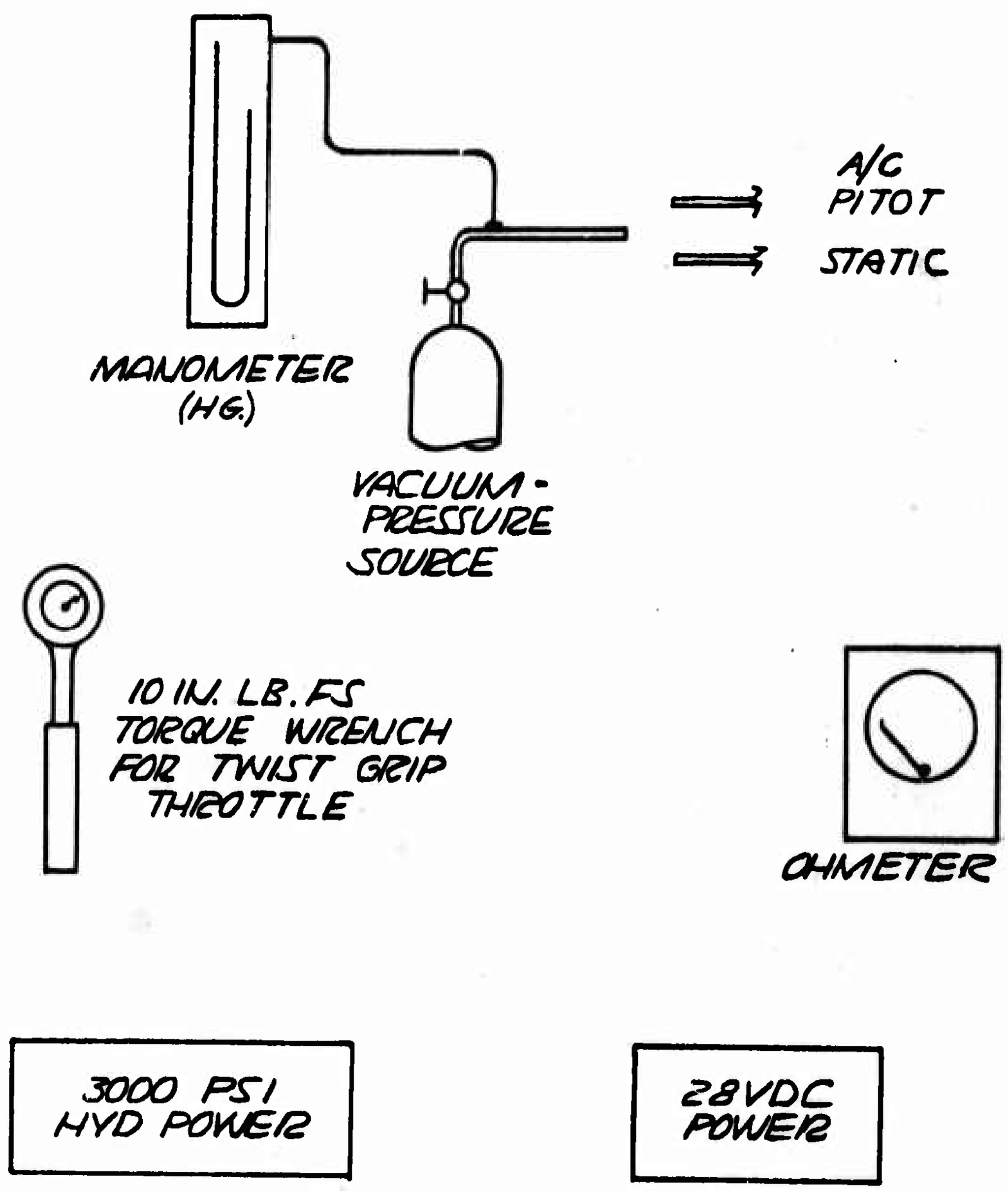

Figure 8 Cockpit Checkout Equipment

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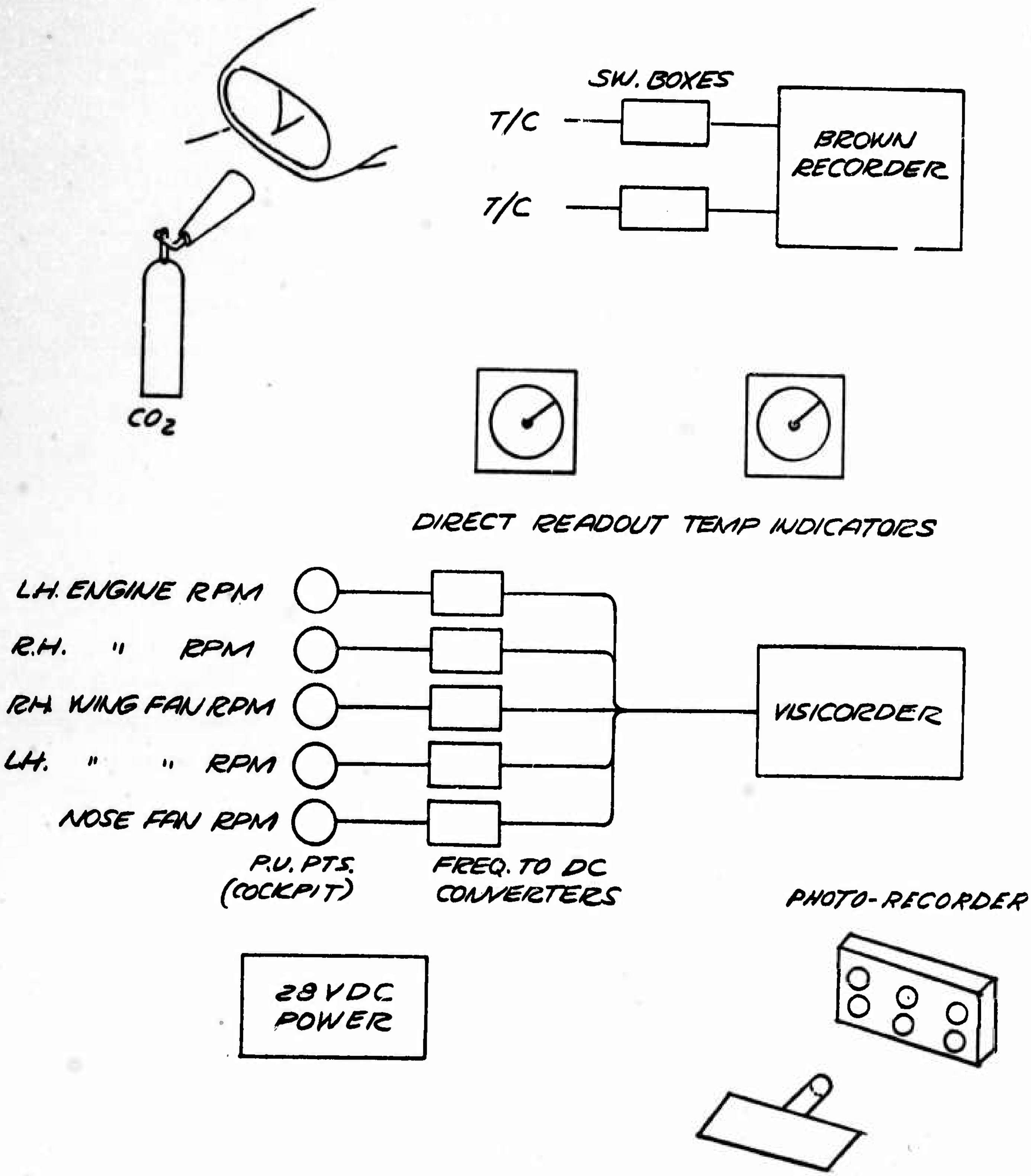

Figure 9 Engine Run and Temperature Survey Instrumentation

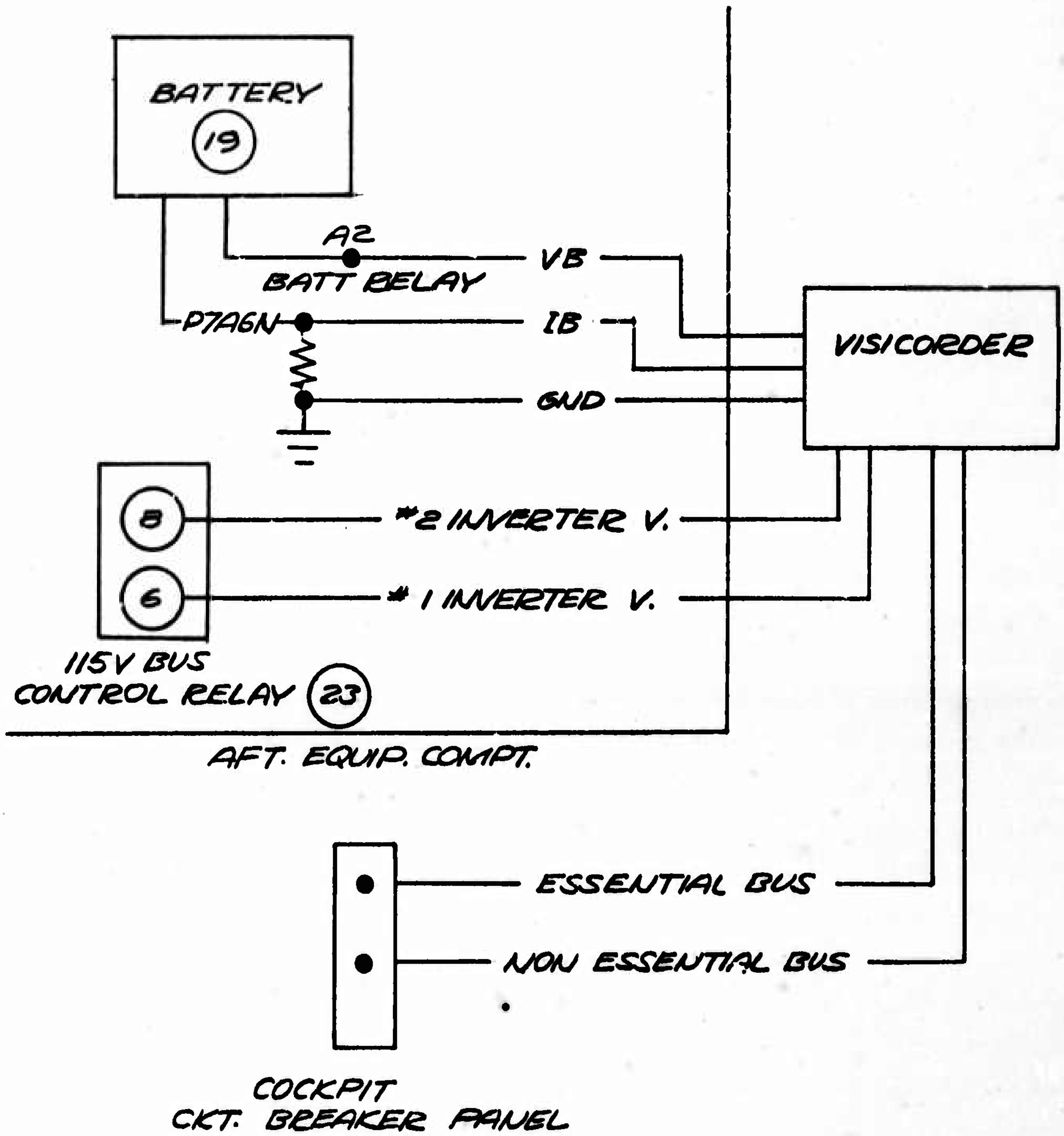

Figure 10 Engine Run Electrical System Checkout Instrumentation

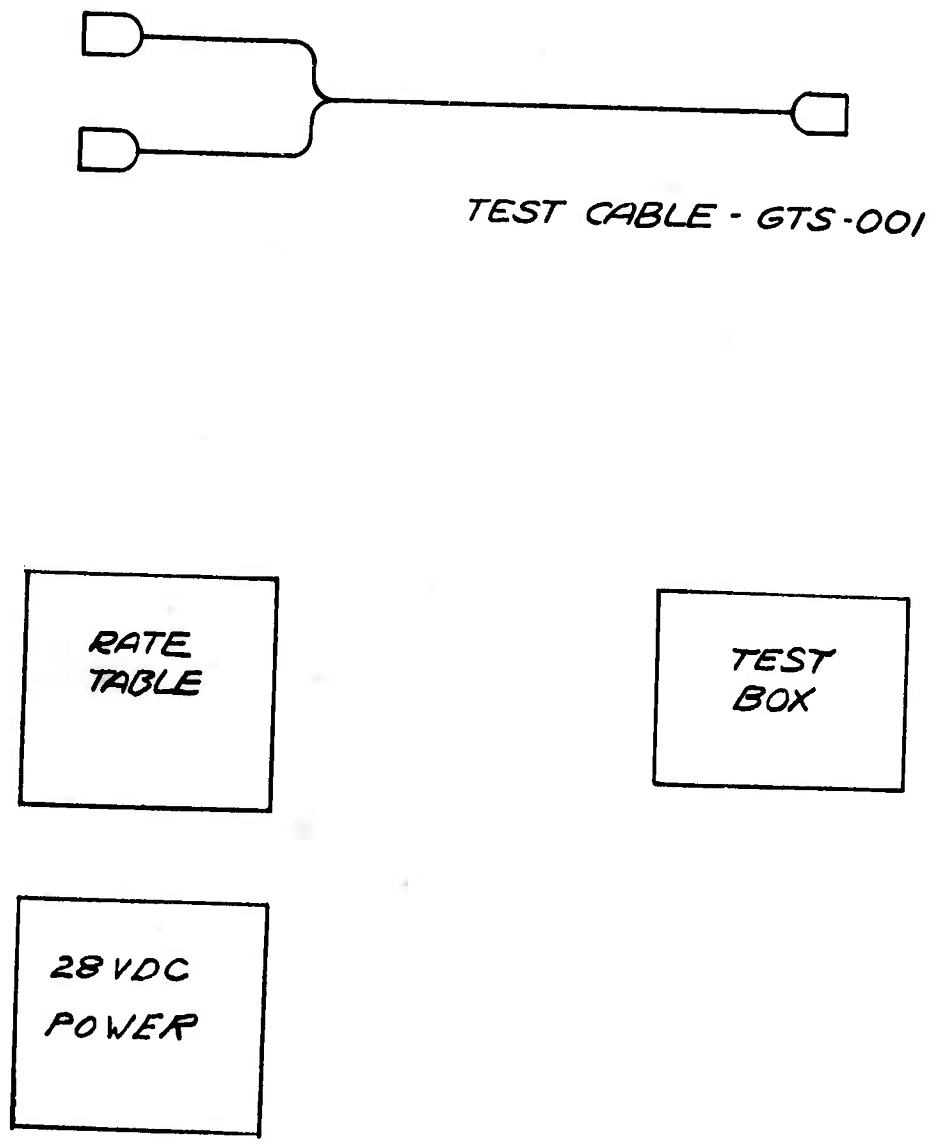

Figure 11 Automatic Stability Amplifier-Gyro Check

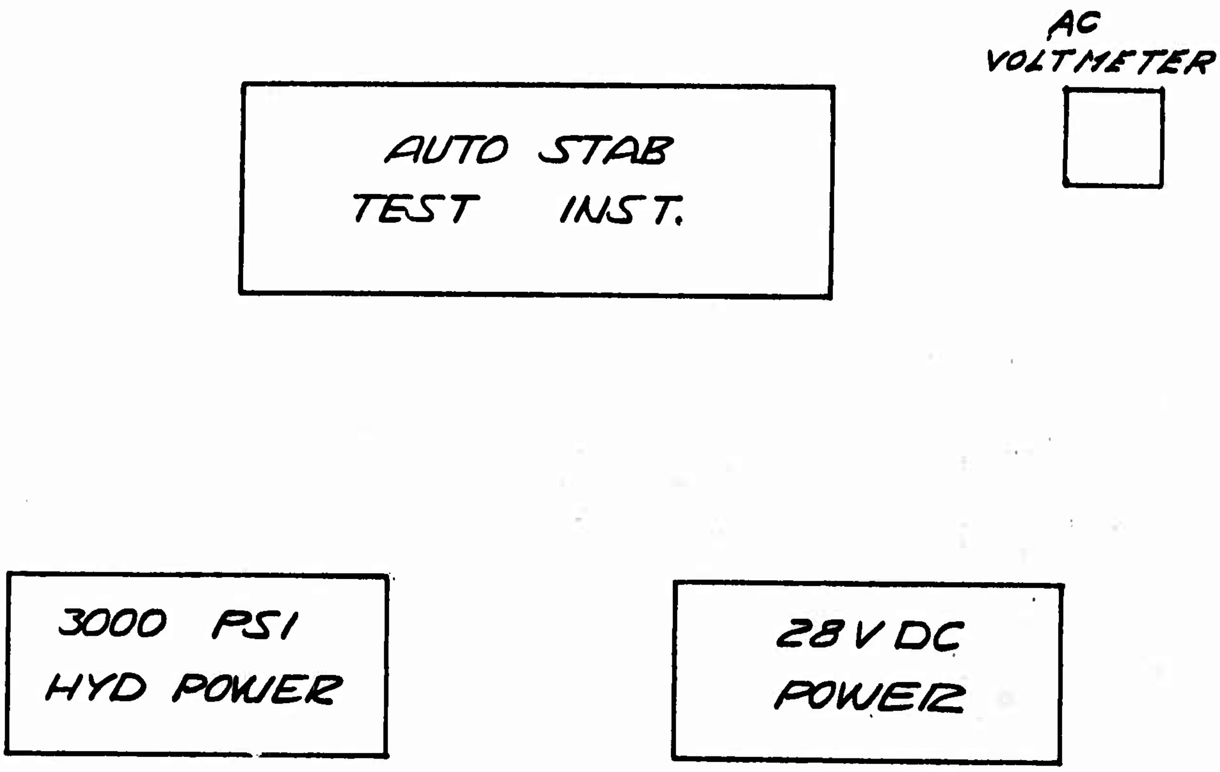

Figure 12 Auto-Stability System Gain Adjustment Equipment

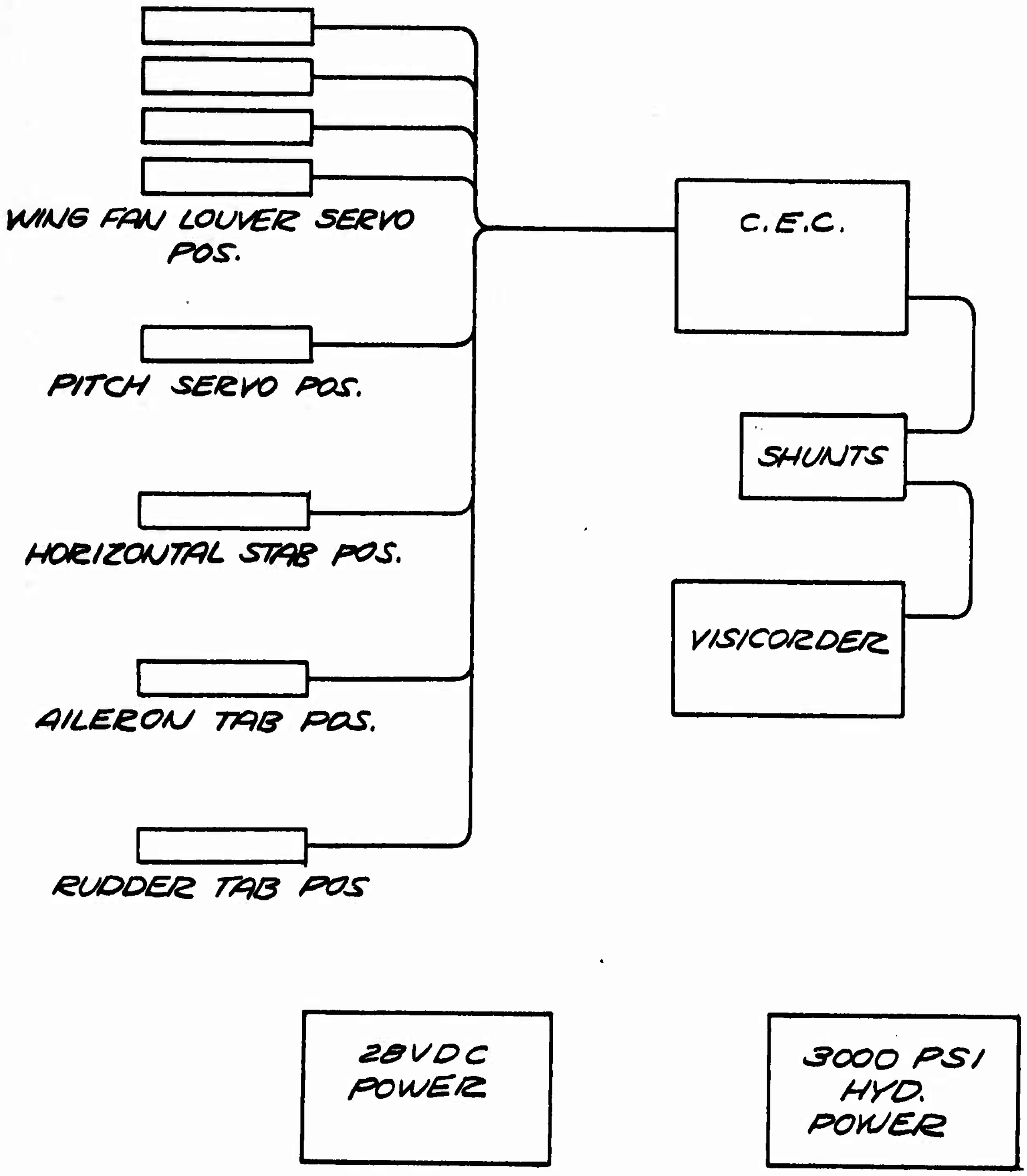

Figure 13 Fan Flight Trim Rates Instrumentation

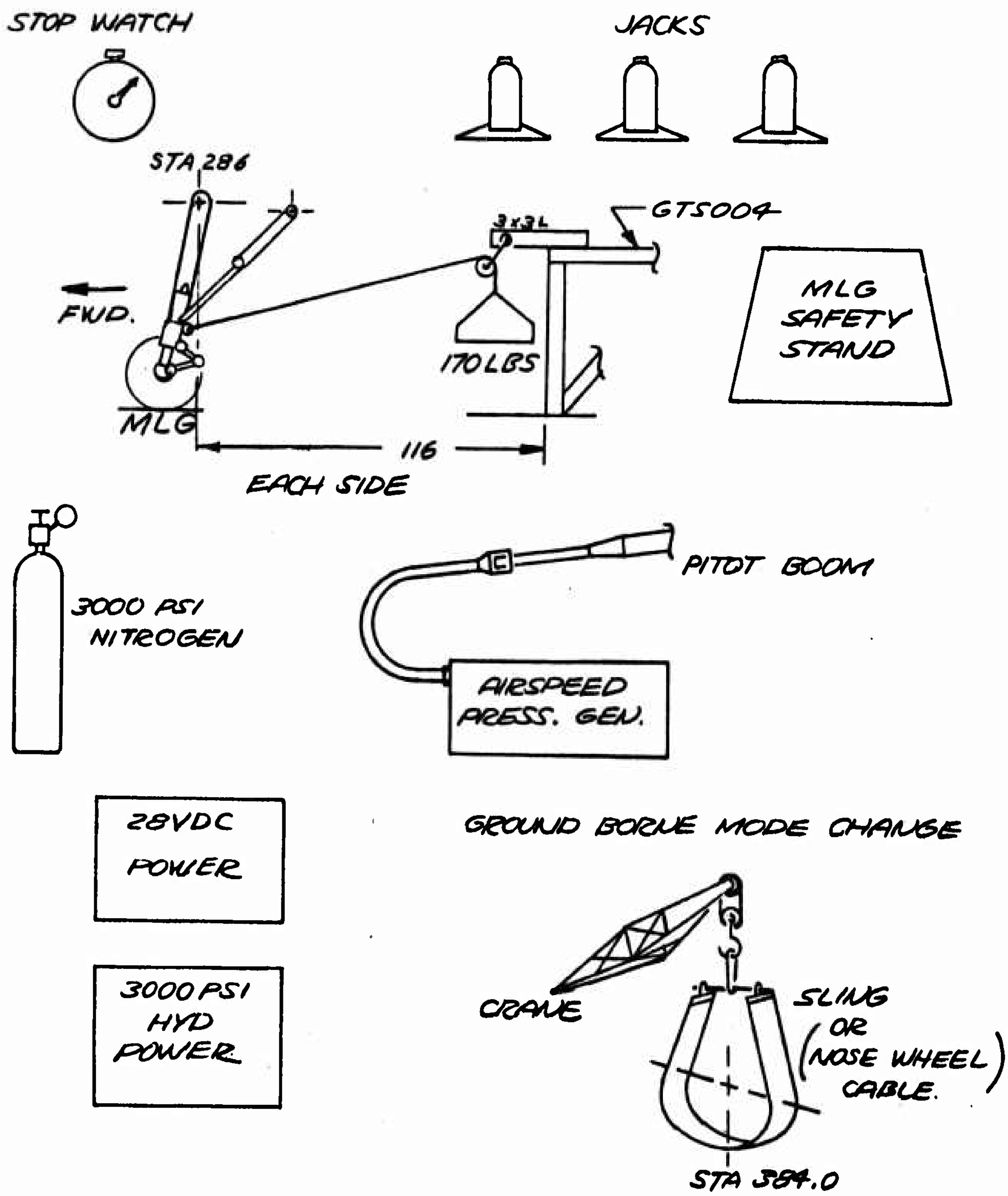

Figure 14 Landing Gear Test Equipment

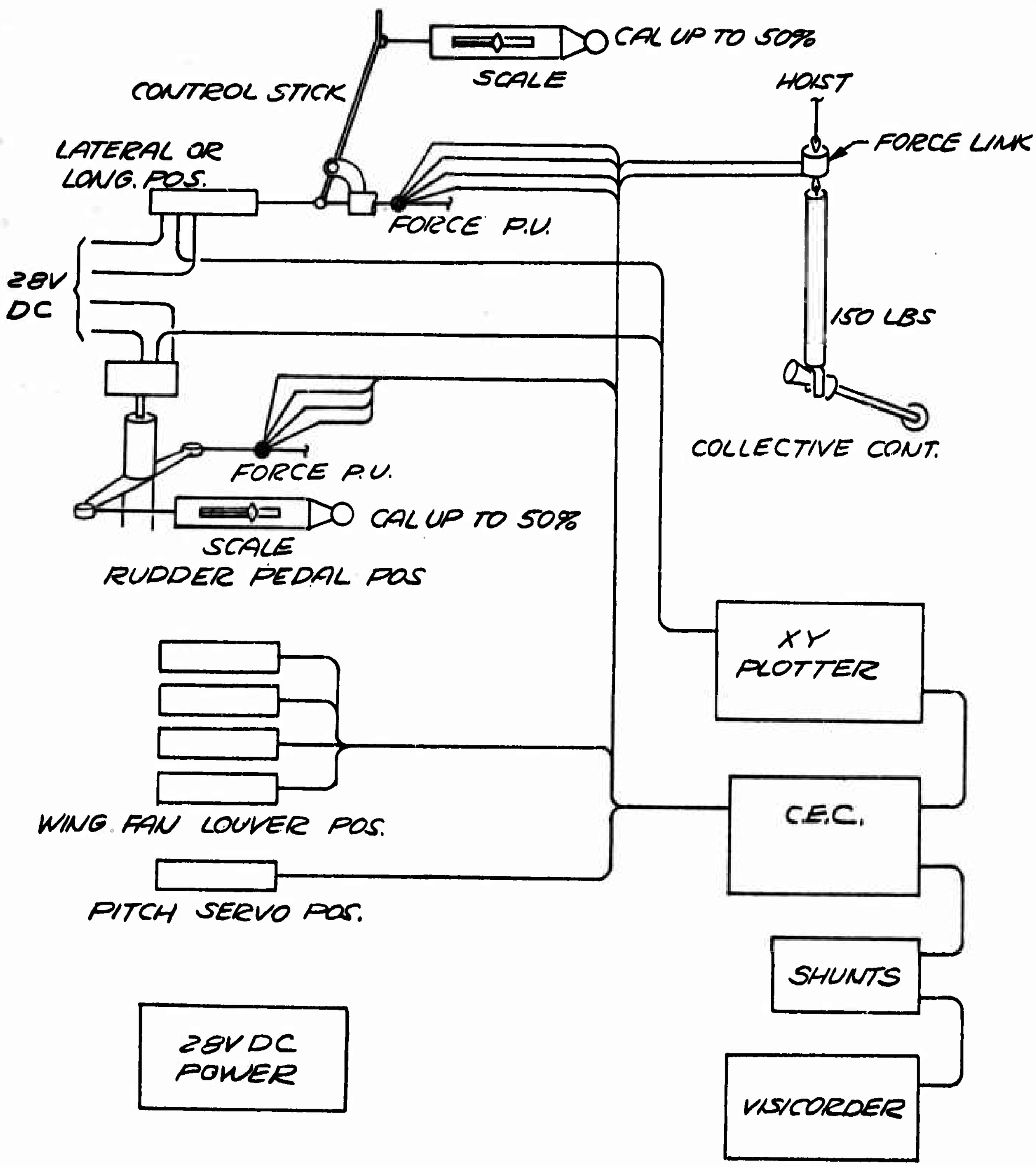

Figure 15 Controls Proof Loads Instrumentation

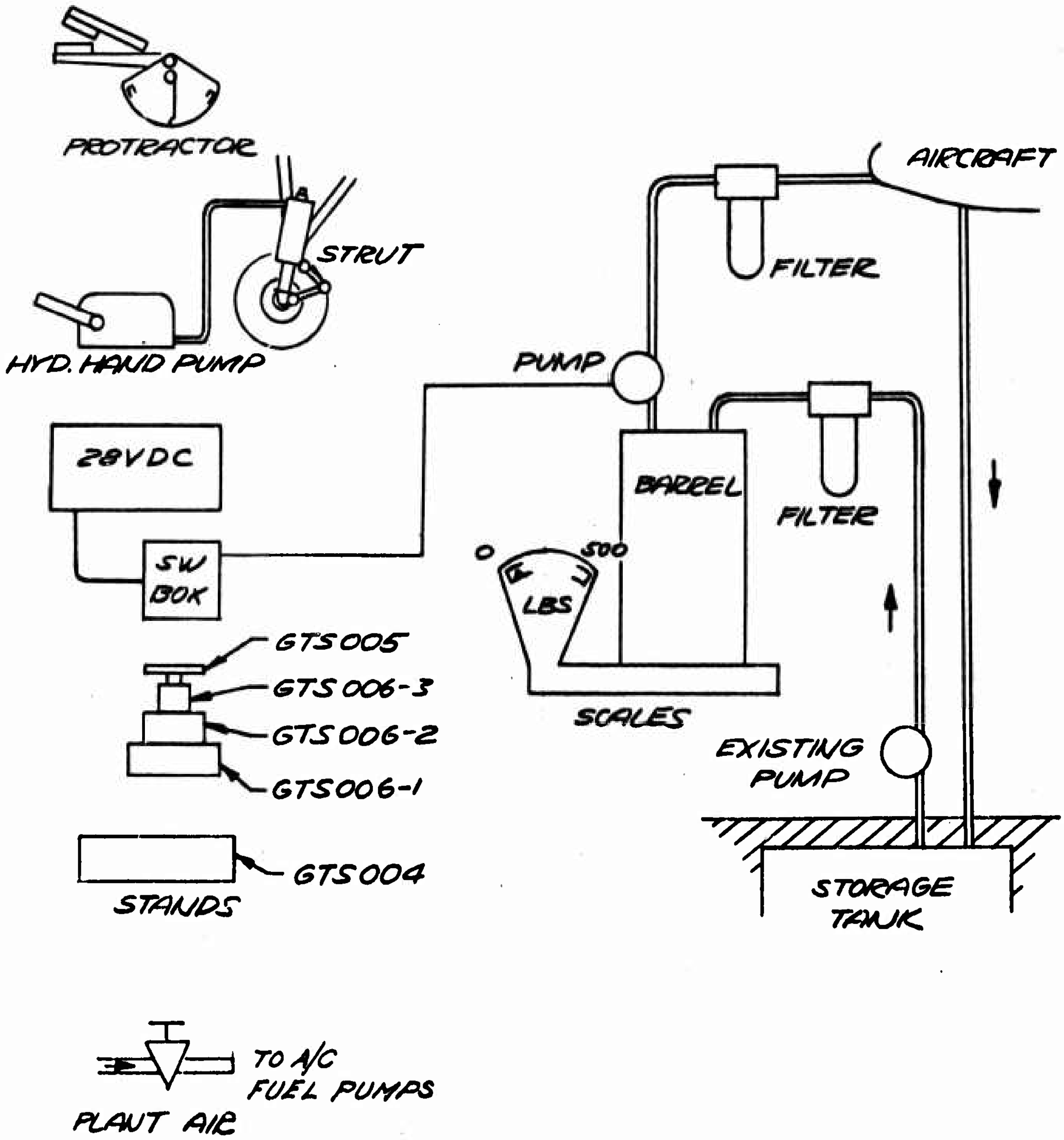

Figure 16 Weights, Balance and Fuel Test Equipment

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A/C ATTITUDE 0°

A/C ATTITUDE	FUEL QUANTITY FWD MAIN LBS ADDED	FUEL QUANTITY FWD MAIN INDIC.	FUEL QUANTITY ATT MAIN & DORSAL LBS ADDED	FUEL QUANTITY ATT MAIN & DORSAL INDIC	A/C WEIGHT NLG (TARE =) LBS	A/C WEIGHT MLG (TARE =) LBS	A/C WEIGHT MLG (TARE =) LBS
0°	0		0				
	200						
	400						
	600						
	800						
	1000						
	1200						
	1400						
	1600						
	FULL						
			200				
			400				
			600				
			800				
			1000				
			1200				
			1900				
			1600				
			FULL				

FUEL LOW LEVEL LEVEL LIGHT EXTINGUISHES:

FWD MAIN INDIC ______ LBS.
INTANK ______ LBS.

AFT MAIN & DORSAL INDIC ______ LBS
INTANK ______ LBS.

UNUSABLE FUEL

FWD MAIN ______ LBS.

AFT MAIN & DORSAL ___ LBS.

Data Sheet 1

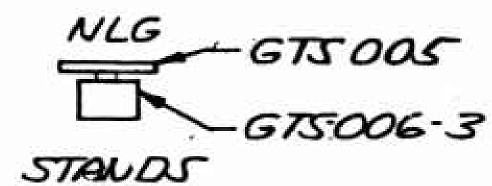
A/C ATTITUDE-5° NOSE UP

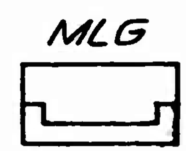
A/C ATTITUDE	FUEL QUANTITY FWD MAIN LBS ADDED	FUEL QUANTITY AFT MAIN & DORSAL LBS ADDED	A/C WEIGHT MLG (TARE:) LEFT	A/C WEIGHT MLG (TARE:) RIGHT
5° NOSE UP	0	0		
	200			
	400			
	600			
	800			
	1000			
	1200			
	1400			
	1600			
	FULL			
		200		
		400		
		600		
		800		
		1000		
		1200		
		1400		
		1600		
		FULL		

UNUSABLE FUEL

FWD MAIN ______ LBS.

AFT MAIN & DORSAL ____ LBS.





Data Sheet 2

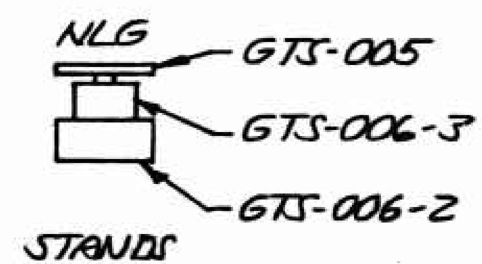
A/C ATTITUDE 10° NOSE UP

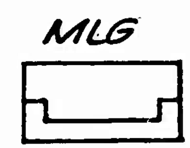
A/C ATTITUDE	FUEL QUANTITY FWD MAIN LBS ADDED	FUEL QUANTITY AFT MAIN & DORSAL LBS ADDED	A/C WEIGHT MLG (GAGE:) LEFT	A/C WEIGHT MLG (GAGE:) RIGHT
10° NOSE UP →	0 200 400 600 800 1000 1200 1400 1600 FULL →	0 → 200 400 600 800 1000 1200 1400 1600 FULL		

UNUSABLE FUEL

FWD. MAIN ______ LBS.

AFT MAIN & DORSAL ______ LBS.





Data Sheet 3

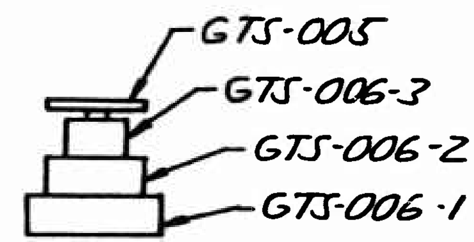
A/C ATTITUDE 15° NOSE UP

A/C ATTITUDE	FUEL QUANTITY FWD MAIN LBS ADDED	FUEL QUANTITY AFT MAIN & DORSAL LBS ADDED	A/C WEIGHT MLG (TARE:) LEFT	A/C WEIGHT MLG (TARE:) RIGHT
15° NOSE UP	0 200 400 600 800 1000 1200 1400 1600 FULL	0		
		200 400 600 800 1000 1200 1400 1600 FULL		

UNUSABLE FUEL

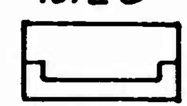
FWD MAIN ______ LBS.

AFT MAIN & DORSAL ______ LBS.



STANDS

MLG



Data Sheet 4

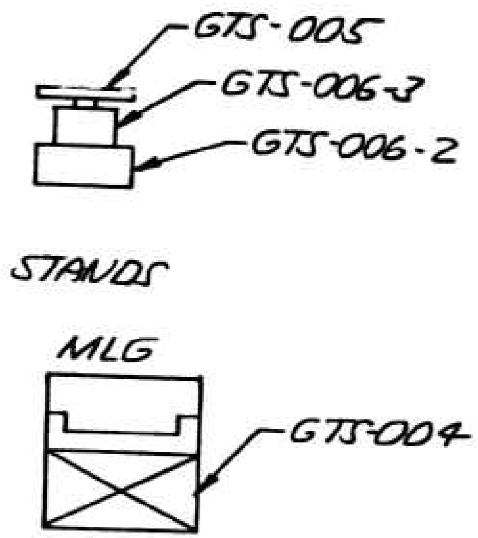
A/C ATTITUDE 5° NOSE DOWN

A/C ATTITUDE	FUEL QUANTITY FWD MAIN LBS ADDED	FUEL QUANTITY AFT MAIN & DORSAL LBS ADDED	A/C WEIGHT MLG (GAGE:) LEFT	A/C WEIGHT MLG (GAGE:) RIGHT
5° NOSE DN	0 200 400 600 800 1000 1200 1400 1600 FULL	0 ↓ 200 400 600 800 1000 1200 1400 1600 FULL		

UNUSABLE FUEL

FWD MAIN ______ LBS.

AFT MAIN & DORSAL ______ LBS.



Data Sheet 5

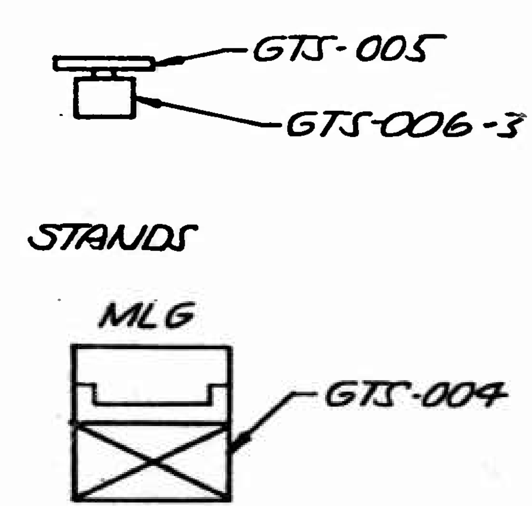
A/C ATTITUDE - 10° NOSE DOWN

A/C ATTITUDE	FUEL QUANTITY FWD MAIN LBS ADDED	FUEL QUANTITY AFT MAIN & DORSAL LBS ADDED	A/C WEIGHT MLG (TARE:) LEFT	A/C WEIGHT MLG (TARE:) RIGHT
10° NOSE DN	0	0		
	200			
	400			
	600			
	800			
	1000			
	1200			
	1400			
	1600			
	FULL			
		200		
		400		
		600		
		800		
		1000		
		1200		
		1400		
		1600		
		FULL		

UNUSABLE FUEL

FWD MAIN ______ LBS.

AFT MAIN & DORSAL ______ LBS.



Data Sheet 6

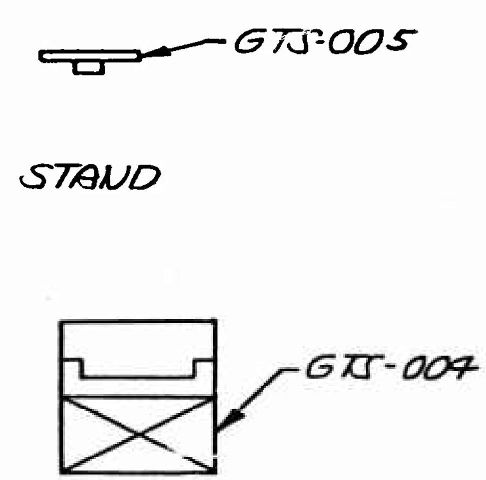
A/C ATTITUDE 15° NOSE DOWN

A/C ATTITUDE	FUEL QUANTITY FWD MAIN LBS ADDED	FUEL QUANTITY AFT MAIN & DORSAL LBS ADDED	A/C WEIGHT MLG (TARE=) LEFT	A/C WEIGHT MLG (TARE=) RIGHT
15° NOSE DN	0	0		
	200			
	400			
	600			
	800			
	1000			
	1200			
	1400			
	1600			
	FULL			
		200		
		400		
		600		
		800		
		1000		
		1200		
		1400		
		1600		
		FULL		

UNUSABLE FUEL

FWD MAIN _____ LBS.

AFT MAIN & DORSAL _____ LBS.



Data Sheet 7

L.H. ENGINE · RUNS

L.H. FUEL FLO	L.H. OIL PRESS	L.H. RPM	LH EGT	#1 HYD PRESS	#1 GEN. VOLTS
		7920 48%			
		10720 65%			
		12390 75%			
		14020 85%			
		14850 90%			
		15700 95%			
		16180 98%			
		16350 99%			
		16500 100%			

R H ENGINE RUNS

R.H. FUEL FLO	R.H. OIL PRESS	R.H. RPM	RH EGT	#2 HYD PRESS	#2 GEN. VOLTS
		7920 48%			
		10720 65%			
		12390 75%			
		14020 85%			
		14850 90%			
		15700 95%			
		16180 98%			
		16350 99%			
		16500 100%			

Data Sheet 8

TABLE I

AUTO-STABILITY
TEST BOX CONNECTIONS AND SWITCH POSITIONS
FOR SYSTEM NULL, PHASE AND GYRO CHECK

Connectors

Connect to Amplifier - 1P2, 1P3, 1P4

Connect to A/C wiring - 1J2, 1J3, 1J4

Switches	Qty.
Ratio - All at "6"	(3)
Pri-Ampli Power - All at "Off"	(3)
Pri-Maneuvering Relays - All at "Maneuver"	(3)
Stby. Maneuvering Relays - All at "Maneuver"	(3)
Load - All at "Out"	(3)
Transfer Switch - "Standby"	
Diverter Valve Interlock - "VTOL"	
Gyro Switch - "Gyro Signal"	
Primary Power Switches - All "Off"	(3)
Stby. Power Switches - All "Off"	(3)
Input Mode - "Gyro"	
Signal - "Low"	
Phase - "Normal"	
Signal Source - "Internal"	
28 VDC - "Aircraft"	
115 VAC - "Aircraft"	
Aux. Input - "Normal"	
Meter Function - As required for test.	
Meter Scale - As required for test.	
Input - As required for test.	
Input Adjust - Not required, adjust to zero	

TABLE II

AUTO-STABILITY
TEST BOX CONNECTIONS AND SWITCH POSITIONS
FOR GAIN ADJUSTMENT

Connectors

Connect to Amplifier - 1P2, 1P3, 1P4

Connect to A/C wiring - 1J2, 1J3, 1J4

Connect as required for test - 3J1

Switches

Ratio - All "6"

Pri. Ampl. Power - All "On"

Pri-Maneuvering Relays - All "Hold"

Stby. Maneuvering Relays - All "Hold"

Load - All "Out"

Transfer switch - "Standby"

Diverter Valve Interlock - "VTOL"

Primary Power Switches - All "On"

Stby. Power Switches - All "On"

Input Mode - "Gyro"

Signal - "Normal"

Signal Source - "Internal"

28 VDC - "Aircraft"

115 VAC - "Aircraft"

Any Input - "Normal"

Meter Function - As required for test.

Meter Scale - As required for test.

Input - As required for test.

TABLE III

ENGINE RUN-TEMPERATURE SURVEY
STRUCTURAL TEMPERATURE LIMITS

MATERIAL	TEMPERATURE LIMIT
Aluminum Alloys	250°F
Titanium	
99 Ti; Fuselage at Thrust Spoiler	550°F
6AL4V Pitch Reverser Doors	700°F
Canted Bulkhead F. Sta. 85	700°F
Magnesium AZ31BH24	
Main Landing Gear Door Skin	250°F
Fuselage Panels	250°F
Steel	
Marage - Space Frame	400°F
Silicone Fiberglas Laminate	
Fan Duct Shroud	700°F
Silicone Rubber	
Aft Fairing - Flap Seal	450°F

TABLE IV

ENGINE RUN-A/C PREDICTED TEMPERATURE RANGES

Region	Estimated Temperature Range - °F
Wheel Wells	
Main	200 - 600
Nose	200 - 400
Main door	200 - 600
Nose door	200 - 400
Landing Gear	
Main	200 - 700
Nose	200 - 700
Wing	
Aft Spars	150 - 320
Fwd Spars	150 - 320
Aft Lwr Panel	200 - 900
Aft Upr Panel	100 - 350
Fwd Lwr Panel	200 - 500
Fwd Upr Panel	100 - 350
Ldg Edge	100 - 350
Aft Fairing	200 - 700
Flap	200 - 800
Fuselage	
Thrust Spoiler	600 - 900
Aft Fairing-Flap	600 - 900
Ldg Edge Wing	150 - 350
Canoe	400 - 900
Space Frame	200 - 400
Pitch Fan	
Well inside	600 - 900
Outside	150 - 350
Thrust Reverser	600 - 900
Extraneous	
Radio Air Intake	(120°F)
Inverters Air Intake	(185°F)

TABLE IV (Continued)

ENGINE RUN-A/C PREDICTED TEMPERATURE RANGES (Continued)

Region	Estimated Temperature Range - °F
Extraneous (Continued)	
Aft Louvers Servos Rod end of barrel	250°F
Thrust Spoiler Act Rod end of barrel	275°F
Diverter Valves (Body)	250°F
Hyd. lines below Cross over duct	200° to 250°F

4.0 APPENDIX

POWERPLANT OPERATING LIMITS

J85-GE-5B

Engine Speed	Steady State (MAX.)	104%
	Transient (MAX.)	108%
	Transient Operation Only	50 - 58%
	Military Speed Fluctuation	±1%
	Idle Speed Setting	47 - 49%
	Idle Speed Fluctuation	±1 - 1/2%
	RPM on Air Starter Power	15%
	O/S Governor in Low Speed Position	95 - 98%
EGT	Starting: 1 second	950° C
	4 seconds	850° C
	11 seconds	750° C
	Steady State Military	680° C
	Steady State Idle	600° C
	EGT Fluctuation	10° C
Vibration	Fwd Compressor)	
	Aft Compressor) ------------	3 mils steady
	Fwd Turbine)	6 mils peaking
Oil Pressure	Idle	5 - 20 psig
	Military	20 - 50 psig
Oil Temperature	Tank	177° C
	#2 Bearing Scavenge	193° C
Oil Consumption		.40 lb/hr
Oil Leakage	Oil Tank Vent at Idle	24 cc/hr*
	Oil Tank Vent at Military	60 cc/hr**
Accessory Drive Gear Box Seal Oil Leakage		
	Fwd. PTO	2 cc/hr
	Aft PTO	2 cc/hr
	O/S Governor Pad	2 cc/hr
	A/B Pump Pad	2 cc/hr
	Main Fuel Pump Pad	2 cc/hr

*Approximately 36 drops in 5 min.
**Approximately 90 drops in 5 min.

Starting Time (Nominal)	20 seconds
Starting Fuel Flow Limit	350 lb/hr max.
Fuel Inlet Temperature	43° C (110° F) max.
Fuel Pressure	-4.7 to 50 psig
Fuel Drainage	
Pump	30 cc/hr
Control	4 cc/hr
Pressurizing Drain Valve	5 cc/hr
O/S Governor	6 cc/hr
Total	45 cc/hr*
Ignition Generator Cycle	2 minutes on, 3 minutes off 2 minutes on, 23 minutes off.
Casing Temperatures	
Fwd Compressor	250° F
Aft Compressor & Mainframe	750° F
Combustor	850° F
Turbine	1150° F
Diverter Valve	1300° F
Thrust Oscillation	±1%
Diverter Valve Actuator Pressure	3200 psi max
Diverter Valve Actuator Oil-in-Temperature	200° F max
<u>X353-5B - Wing Fans</u>	
Fan Speed	103% max
Fan Vibration (Bulletnose vertical)	10 mils steady 20 mils transient
Fan Bearing Temperatures	350° F max
Compressor rotor (turbojet mode)	250° F max
Front frame (outboard side)	300° F max
Exit Louver Actuator Pressure	3200 psi max

*Approximately 70 drops in 5 minutes.

X376 - Pitch Fans

	Fan Speed	110%
	Fan Vibration	15 mils
		8 g's
Fan Temperatures		
	Front Frame	250° F max
	Bearings	350° F max

Ground Check-Out in Fan Mode

	Time	Engine RPM	Vector Angle	Purpose
(a)	5 min.	<86%	-10° to 45°	Check-Out
(b)	1.5 min.	100%	0°	Check-Out

RYAN 63B102